Aeronautical Engineering and Airplane Design

Klemin

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by

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PREFACE

In submitting the series of articles appearing in Aviation and Aeronautical Engineering in book form, only minor corrections have been made.

No attempt has been made for obvious reasons to include new material at hand, and under stress of urgent war work no systematic revision has been attempted.

It is felt, nevertheless, that as the articles contained matter mainly regarding fundamental principles, that they will still be of assistance, particularly to younger designers and draftsmen, while they should be of value as a reference to more experienced men.

The author has, unfortunately, not had the advantage of Mr. Huff's valuable collaboration in this production in book form. He thanks Mr. G. M. Denkinger, instructor in aeronautics at the Massachusetts Institute of Technology, and Mr. Clarence D. Hanscom for valuable assistance in corrections.

Alexander Klemin.

Dayton, Ohio,
October, 1918.

INTRODUCTION

This work, practically a course in aerodynamics and airplane design, is subdivided into two parts: Part I, Aerodynamical Theory and Data; Part II, Airplane Design.

In PART I it is proposed to deal briefly with the fundamental ideas and theories of aerodynamics in a simple yet comprehensive manner.

It is important for the aeronautical engineer and for every student of aerodynamics to have at his disposal exact definitions of such terms as lift, drag or resistance, center of pressure, wing cord, angle of incidence, and other well known expressions.

Although the exact nature of viscosity, skin friction, eddying or density resistance, stream line flow, turbulent flow, the sustaining action of cambered wing surfaces, and the principles of comparison for forces on bodies of varying dimensions still present many difficulties, it is hoped to give a simple and, above all, practical summary of these points. The more difficult theoretical demonstrations will be reserved for special articles.

The authors propose also to give a brief description of the chief aerodynamical laboratories and of experimental methods there employed. Without a knowledge of such methods, appreciation, and application of the laboratory data available is certainly not easy.

Considering the comparatively recent growth of aerodynamics, the amount of material now available is extraordinary. It is unfortunately scattered through a variety of publicatons; English, French, German, Russian and Italian, presented in varying ways and in varying systems of units. Nor is all of it entirely worthy of credence.

In this course it has been attempted to reduce this material, particularly that of English and French origin, to one system of presentation with forces measured in pounds, areas in square feet and velocities in miles per hour or feet per second, so as to be more readily applicable in American design; to include all the material which is trustworthy and of immediate and pressing utility to the designer, in carefully classified form.

The Economic Laws of Flight will be fully dealt with, in horizontal and ascensional flight. The consideration of the performance curves of a machine will be particularly useful to those engineers and students to whom the subject is comparatively new.

Throughout, illustrative problems will be worked out on important points, especially to facilitate comparison between wing sections.

PART II will include a discussion of available aeronautical materials, timber, steel, alloys, rubber, etc.—with trustworthy values for stresses; a variety of diagrams and scale drawings representative of modern design, and a classification of the most important modern machines, with their main data.

At this stage of the art, it is impossible to say that any method in design is standard, but a systematic procedure of design will be fully developed.

Particular stress is laid on the evaluation of factors of safety. The dynamic factor of safety, the material factor of safety, the worst loading possible in the air, the worst possible shock on landing; nothing offers so many possibilities of confusion and untrustworthiness; and nothing is in more need of definite and accurate statement.

Complete strength calculations will be presented for body, chassis, wing girders, and controlling surfaces, and the design of a standard machine will be carried through, with consideration of motor and propeller problems, weight distribution and balancing.

Throughout the course, the most elementary mathematics are employed, and nothing beyond a knowledge of the first mechanical principles is presupposed.

It is hoped, therefore, that the course will be easily understood by any engineer or student approaching the serious study of the airplane for the first time. At the same time it is felt that much will be of service even to the expert aeronautical engineer.

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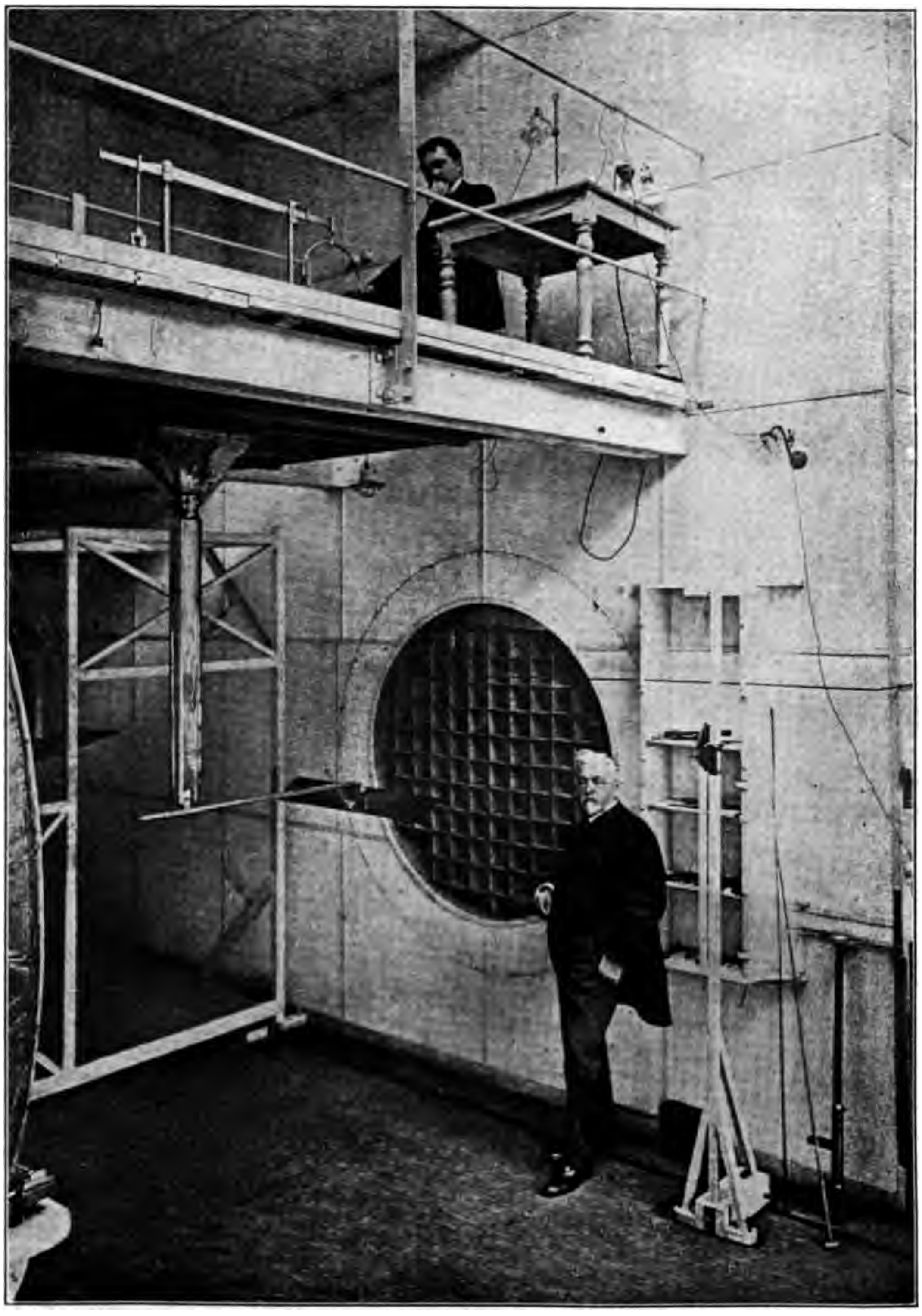


M. EIFFEL IN HIS LATEST AERODYNAMICAL LABORATORY

Chapter I

Modern Aeronautical Laboratories

Early Experimental Aerodynamics

Aeronautics as a whole and aviation, the science of the heavier than air machine, has from its earliest conception, been an experimental art. When Professor Langley in 1887 started his experiments on an extended scale for determining the possibility of, and the conditions for, transporting in the air a body whose specific gravity is greater than that of air.

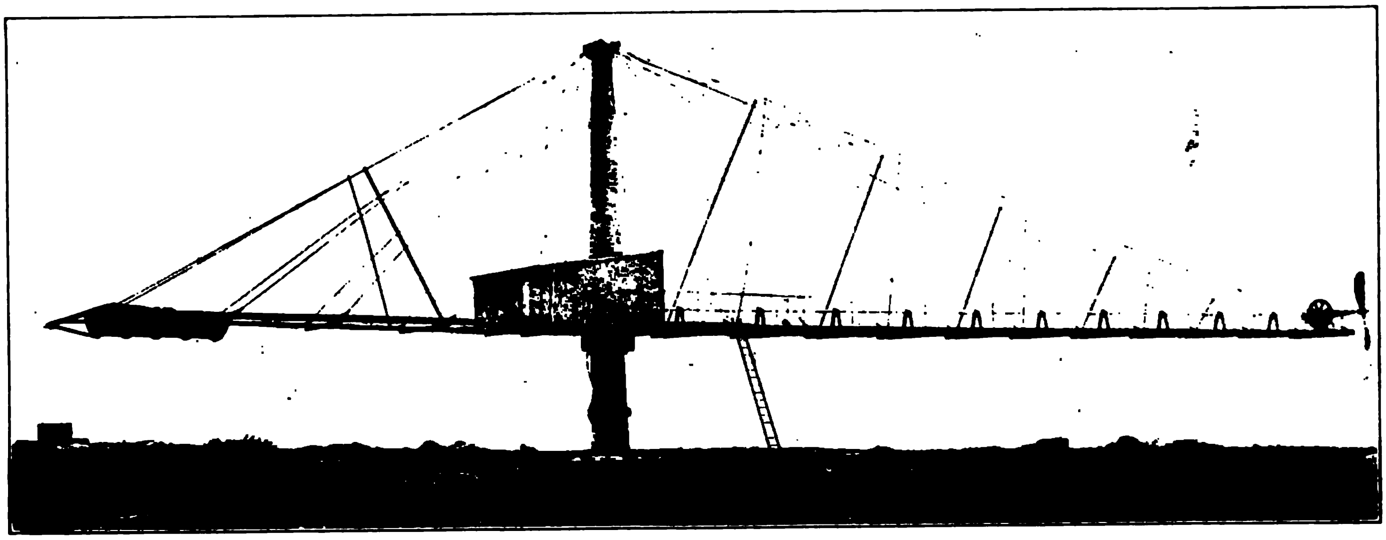


FIG. 1. WHIRLING ARM USED BY MESSRS. VICKERS IN TESTING PROPELLERS

he had before him papers by such scientists as Gay-Lussac and Navier, proving conclusively that mechanical flight was impossible.

Langley was not easily discouraged and by a carefully conducted series of experiments carried on under very adverse conditions, he was able to build a machine which though unsuccessful in its flight in his day, due to faulty mechanism in the launching device, has since been flown under its own power by Glenn Curtiss in 1914, at Hammondsport, N. Y.,—possibly with some alterations.

At the time the Wrights took up the subject in 1896, there were but few aerodynamical works of interest or value in existence. They were dependent upon the meager experiments and tables of Lilienthal and Duchemin and the work of Langley which seemed to verify Duchemin's formula. After spending two years experimenting upon these figures of Lilienthal and Duchemin, the Wrights came to the conclusion that the tables were so much in error as to be of no practical value in airplane design.

In 1901 the Wrights designed and built a small "Wind-Tunnel" in which they could carry on systematic investigations on the pressure produced by various surfaces when presented to the air at different angles. The instruments used in measuring these forces were designed with the intention of correcting the errors which had rendered so untrustworthy the results of their predecessors.

During the winter of 1901-1902 their investigations included some hundred different surfaces of which about half have been tabulated and the results used in their subsequent work. Experiments were made on the effect of varying aspect ratio, curvature, camber, and the variation of the position of the maximum ordinate of the wing section from the leading edge. Thick and thin surfaces were tested to determine the effect of thickness. The effect of superposing the surfaces, as well as placing one behind the other, were measured and what was of even greater interest, the first measurements of center of pressure motion on curved surfaces at carrying angles were tabulated by them. As a direct result of their laboratory experiments and the development of a system of control, worked out in their earlier gliding flights, they were able to build the first power driven airplane.

To demonstrate that the United States deserved a right to leadership in aviation in the earlier years, one need but mention other names, such as those of Octave Chanute and Dr. Zahm. The latter, through the efforts of Hugo Matthullath, was provided with an aerodynamical laboratory which was in its day the most perfect of its kind; and although the experiments extended over a few years only, the results of Dr. Zahm's labors were exceedingly valuable.

General Requirements in Airplane Design

As is the case in ship building, a suitable machine for every purpose cannot be developed and there must be a special type with specific qualities in slow speed, high speed, weight, armament and defense. Some of these factors are directly opposed to others. For example, the ideal machine for the regulation of artillery fire, would be able to remain immovable or circle about very slowly above one point. The "chaser" or machine used to rid the air of the enemy's planes should be the fastest possible. With the comparatively narrow range of speed possible in an airplane one can see the uselessness of an attempt to combine these two types in one machine. On the other hand from the productive side, it is impracticable to increase the number of types indefinitely, for this would call for an enormous outlay in machinery and increase in personnel. A compromise has therefore been made, with the selection of some four master types of airplanes which may be classed according to their military uses:

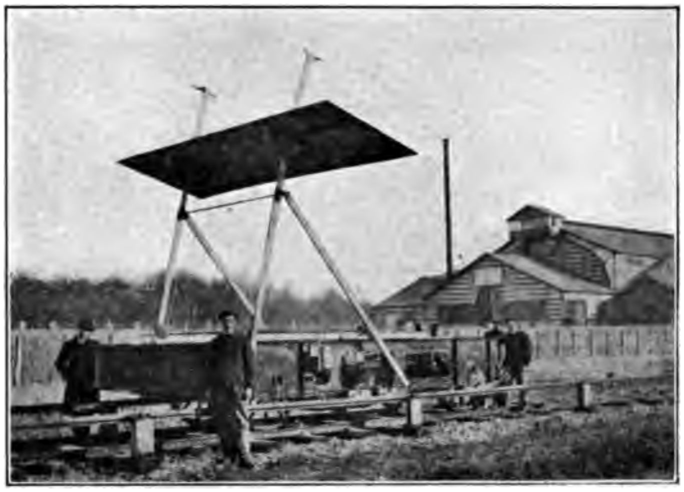


FIG. 2. FIRST PLATFORM EQUIPPED FOR TRIAL AT AEROTECHNIC INSTITUTE OF SAINT-CYR

1. THE STRATEGIC SCOUT. A slow endurance machine for use on long raids into the enemy's country, for mapping and photographic work.

2. THE HIGH SPEED SCOUT. For tactical reconnaissance and use over the lines, and capable of out-climbing and out-flying the enemy.

3. FIGHTING OR BATTLEPLANE. Armed and armored, for driving off the enemy's scouts and protecting the fourth class.

4. BOMB DROPPERS OR WEIGHT CARRIERS. For use in destroying small bridges, railways, etc., depending for their protection upon the battleplane.

In order to design and build machines to meet such qualifications the designer must give up the old haphazard methods of building first, and then determining the performance. He must go about the design in a thorough and scientific manner in order to hope to come within reasonable limits of his specification.

The most important items in the performance of present-day machines are: their weight, their rate of climb, high and low speeds, angle of glide, propeller efficiency, and endurance at economical speed for various loadings. These depend on a careful manipulation of aerodynamical data, including the lift and resistance of the main planes and control surfaces, the resistance of struts, wires, wheels, radiators and appendages, the distribution of loads on surfaces, and different combinations of surfaces. On the effects of the various supporting, control and fin surfaces, and on the summation of all aerodynamical forces depend not only the performance, but the controllability, factor of safety, and stability of the airplane. To produce a desired type the designer must bear in mind every factor.

The desired type can be obtained by the "cut and try" process on the full size machine. This experimental flying is, however, a dangerous and costly method that has led to many an unfortunate accident.

Difficulties of Full Scale Experiments

The real worth of full scale experiments depends on the delicacy and precision of the recording instruments, the expertness of the pilot and the interpreter of the recorded data. The chief objections, other than that of danger to the pilot, are the great variations in atmospheric conditions and therefore the unavoidable delays in tests, the inability to repeat the trials under exactly the same conditions, the necessarily short time allowable for observation and the unavoidable introductions of many variables, when but a slight change is made in one part of the design. It is this inability to discriminate among the possible causes of behavior of the machine that may lead to a maze of conflicting results.

There is a place, nevertheless, and a very important one for full sized experimental flying—that the machine may be tuned up and minor adjustments made for ease of control and steadiness under actual flying conditions. Such work, however, should not be undertaken until the safety of the pilot is reasonably assured.

Towing Methods

The most natural and logical thing to do with model airplanes would be to tow them through still air and record the forces and moments to which they are subjected. This is not so simple an arrangement as in marine work. The airplane is free to move along the three axes in space and around any of the same, which introduces complications in the recording mechanism that are most difficult to overcome. Very much higher speeds are required in aeronautical work and this increases the length of track for testing prohibitively or decreases the time of experimental observation to such an extent as to spoil the precision of the results.

The principal objection to towed model flight is the inability to obtain *still* air, as even in a closed room eddies are constantly present, which are impossible of measurement; this may be observed by making apparently calm air visible by the introduction of smoke. Radiation of heat from the walls is apt to cause such eddy making to a very marked degree.

In a measure the difficulties of rectilinear-motion are overcome by replacing it by rotation about a fixed axis; but here the radius must be relatively large and the building necessarily of similar great dimensions. The rotation is not wholly comparable with translation since along the transverse axes of the body, under test, the different parts have not the same relative velocity and some compromise is necessary due to this difference in radial length. Centrifugal force is present which must be overcome by the measuring instrument, as well as the disturbance set up in the air by so large an object as the whirling-arm passing the same point a number of times.

The whirling-arm used by Messrs. Vickers, Ltd., of England. in their experimental work is illustrated in Fig. 1.

Wind Tunnel Methods

If we are willing to accept the doctrine of relative motion, then the resultant force on a solid with a uniform motion through still air, is the same as that for an immovable solid upon which a constant current of air impinges. A "Wind Tunnel" test, where a steady current of air impinges on a model at rest, should therefore give the same results as a towing test. Differences would be due to experimental errors and not to a difference in principle.

In the towing method, the influence of the mounting stage and unsteadiness of the air introduce errors. In the wind tunnel, there may be slightly non-uniform flow, disturbances due to the sides of the tunnel, etc. Wind tunnel work, however, has proved far superior to the towing method, which it has almost entirely replaced and it has now been developed to a high degree of precision and usefulness.

From wind tunnel tests, the engineer may obtain data for the "balancing" up of an airplane—the adjustment of the center of gravity with reference to the air forces, the loading on his wing and control surfaces, the resistance of the body and appendages, and other useful information. It will be scarcely disputed that such tests are of immediate commercial value to the practical designer.

Laboratories of the Wind Tunnel Type

The Institut Aérotechnique de l'Université de Paris, under the directorship of M. Maurain and M. Toussaint, situated at St. Cyr, some ten miles out of Paris, is devoted, for the most part, to experiments on full size surfaces and aeroplanes. Covering some eighteen acres of land, a splendid opportunity is offered for ample buildings, as well as the seven-eighths of a mile railway track used for experimental work.

The main building with a large central hall is surrounded on three sides with work shops, laboratories and a power station. Within the hall is installed the experimental apparatus directly connected with aviation. Here there are several wind tunnels of different dimensions and wind speed, arranged for the testing of scale models and appendages, apparatus similar to Colonel Renard's for the investigation of stability and proper propeller testing apparatus. A motor testing plant for endurance and economy of aeronautical motors, instruments for measuring the propeller torque for various rotational speeds at a fixed point and the testing of propellers at rupturing speeds are also included.

In the chemical laboratory investigations on balloon fabrics and gases are undertaken with special reference to their manufacture and purification. The physical laboratories are devoted to the production of instruments for aeronautical purposes, both experimental and applied. Work shops are at one end and an individual power station supplies energy and light to the Institute and experimental departments.

In a separate building, covering a quarter of an acre, is housed a "whirling-arm" some 50 feet in radius, used principally for the calibration of instruments. It is, however, not as popular as the track and wind tunnel experimental apparatus. The out-door track proper is of standard gage, seven-eighths of a mile long, level for the part over which experimental data is recorded, but rising slightly for some distance at either end to facilitate the starting and stopping of the five-ton electric car, upon which the surface, full size airplane or propeller is mounted and carried.

Four cars, each rigged for one type of experiment, are considered necessary. Car number one measures the horizontal and vertical components of the resultant air force, as well as the center of pressure for various angles of incidence of the surface to the wind; two and three are for large and small propellers in connection with dirigible and airplane work; and number four is especially equipped for the measurement of the resistance of appendages. The carriages are equipped with appropriate measuring instruments of the recording type, readings being recorded simultaneously as the car moves over the track. The velocity of the air is recorded by means of a calibrated venturi tube anemometer.

The testing apparatus is of such size that a full scale airplane can be mounted and subjected to test for lift and resistance and static longitudinal stability. In order to accommodate such large forces as are encountered in full scale work, the instruments are of considerable size; this has the disadvantage of destroying much of the delicacy of the measurement. The results obtained are said to be in error by about five per cent in lift and about ten or fifteen per cent in resistance.

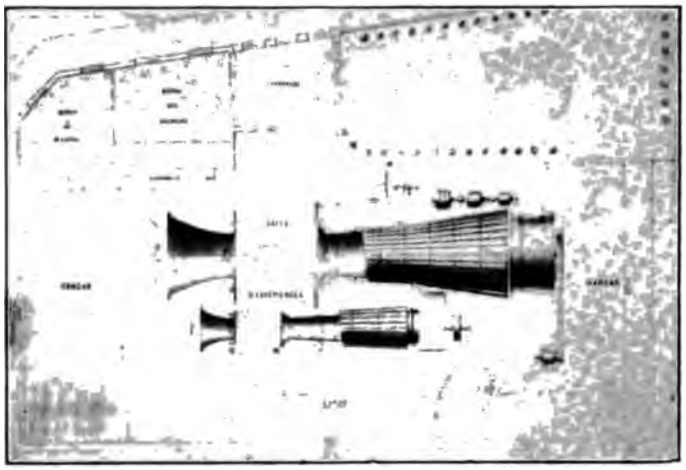


FIG. 3. PLAN OF EIFFEL'S AERODYNAMICAL LABORATORY

The real value of experimental work of this nature and its comparison with results obtained from model tests is as yet not fully determined.

Eiffel in his laboratory at Auteuil, in an investigation to ascertain the aerodynamical effect produced by the car, situated as it was directly beneath the surface under test, reports that when correction is made for the presence of the carriage, agreement with wind tunnel results is fairly obtained. He also notes that modification is being made at St. Cyr in the position of the surface as mounted on the car in order to reduce the interference as much as possible. A similar comparative test made at the National Physical Laboratory, England, on wind tunnel models shows good agreement with the lift coefficients, but the resistance coefficient in the full size experiment is still unsatisfactory.

The Laboratoire Aérodynamique Eiffel, supported by the personal means of, and directed by G. Eiffel, is of the most elaborate in design. Devoted entirely to wind tunnel experiments, it is completely housed in a beautiful white stone building, fronted by a formal garden. The building proper, two stories in height, is 100 by 40 feet.

As may be seen by the accompanying photographs, the laboratory room is rectangular in shape, with a large and small wind tunnel side by side, occupying the central space and suspended mid-way from floor to ceiling. The position of the tunnels permits the free circulation of the air in the room. The wind tunnels are of similar character, one being of smaller working diameter than the other. They each consist of a bell-shaped collector, a large laterally *air-tight* experimental chamber, used for both the large and small tunnel, and independent expanding trunks leading from the experimental chamber to the individual suction blowers. The air is drawn from the large surrounding room or hangar into the bell collector, through a honey-comb baffle to straighten the flow, then across the experimental chamber into the expanding trunk where it passes through the suction blower and is discharged at low velocity, back into the room.

M. Eiffel's characteristic variant is his hermetic experimental chamber. When first interested in experimental aeronautics, he experienced difficulty, due to interference of flow around his models caused by the walls of the closed tunnel. In order to avoid a tunnel of excess size and still not reduce his model dimensions, the walls were removed for some distance and replaced by an air-tight chamber enclosing the stream of air. The pressure in the hermetic room is necessarily that of the air stream it contains, so that a cylinder of air traverses the chamber in parallel stream lines and without showing any appreciable eddy. If a fine silk thread is held in the working stream, a slight play up and down or to the right and left may be noted, showing some variation is present. The velocity of the stream is measured by an alcohol manometer, registering the difference in pressure in the experimental chamber and the laboratory room outside. This is one method of velocity determination and will be explained in detail later. The manometer when left to itself shows a slow variation in velocity with time of some four per cent.

The general dimensions of the installation at Auteuil are as follows: The large channel has a bell collector with end diameters of 4 and 2 meters, 13.1 and 6.6 feet, with a length of 3.3 meters, 10.8 feet, and an expanding trunk 9 meters, 29.5 feet, long with end diameters equal to the collector. The expanding trunk connects with a suction blower, having a sectional area of 9 square meters, 97 square feet. The small tunnel comprises a bell collector, ends 2 meters and 1 meter in diameter, 1.65 meters in length and an expanding trunk 6 meters long, connected to a Sirocco suction blower.

The above dimensions permit in the larger, a uniform stream of air 2 meters, 6.6 feet, in diameter to be drawn through the experimental chamber at a speed varying between 2 and 32 meters per second, or 6 and 105 feet per second; this is accomplished by a 50 h.p. electric motor driving a 50 per cent efficient blower. In the small tunnel of 1 meter diameter air flow, a maximum velocity of 40 meters, 131 feet, per second, is obtained by a 50 h.p. electric motor driving the Sirocco blower.

The experimental chamber is a rectangular room free from obstructions 10 meters, 33 feet, in length, 4 meters, 13.1 feet in width and 5 meters, 16.4 feet, in height. A rail supporting a sliding floor carries the observer and weighing mechanism above and clear of the air stream. A second observer on the floor is required to regulate the wind and adjust the model during a test. The two tunnels are of course so arranged that the one not in use may be blocked off with air-tight wall plates so preserving the low pressure in the experimental chamber. In order to avoid the possible physical discomfiture often accompanying sudden changes in pressure, an air-lock is provided for passage into or out of the experimental chamber.

The models are mounted upon especially designed standards or measuring instruments, such as a large and small aerodynamical balance—devices for measuring the lateral forces, pressure distribution and a very excellent apparatus for testing small propellers. All the apparatus is mounted in the most convenient manner and may be used for either the large or small channel as desired. The accuracy of the results obtained, while possibly not sufficient for exact physical research, are ample, from the practical stand-point.

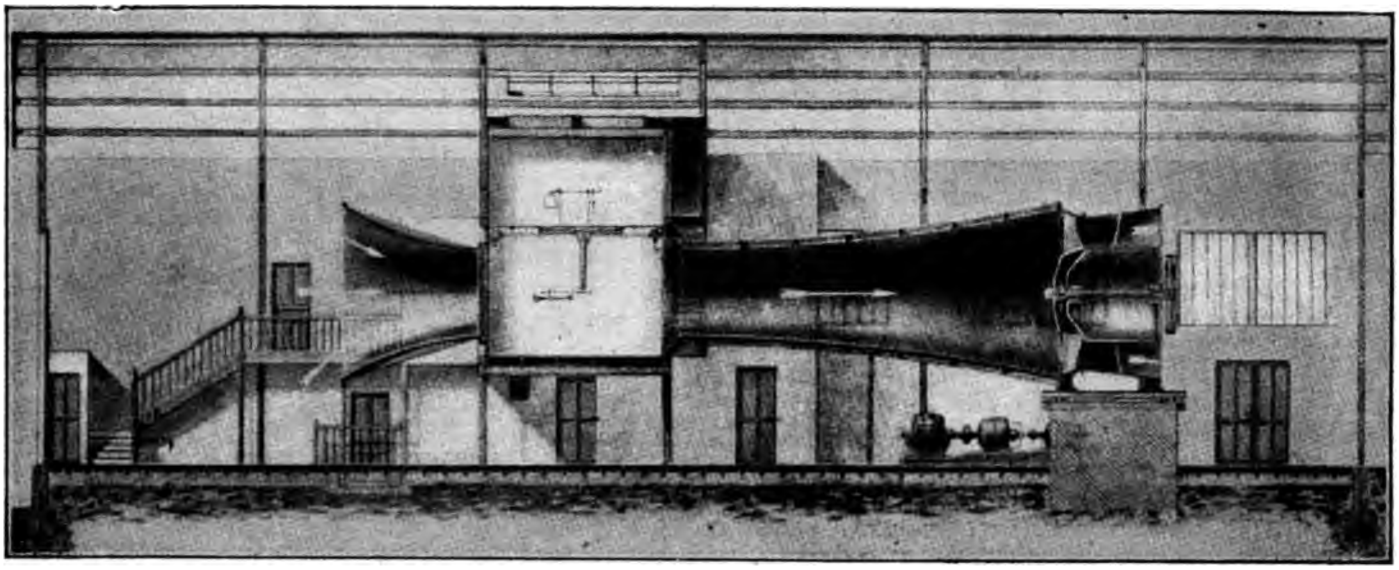
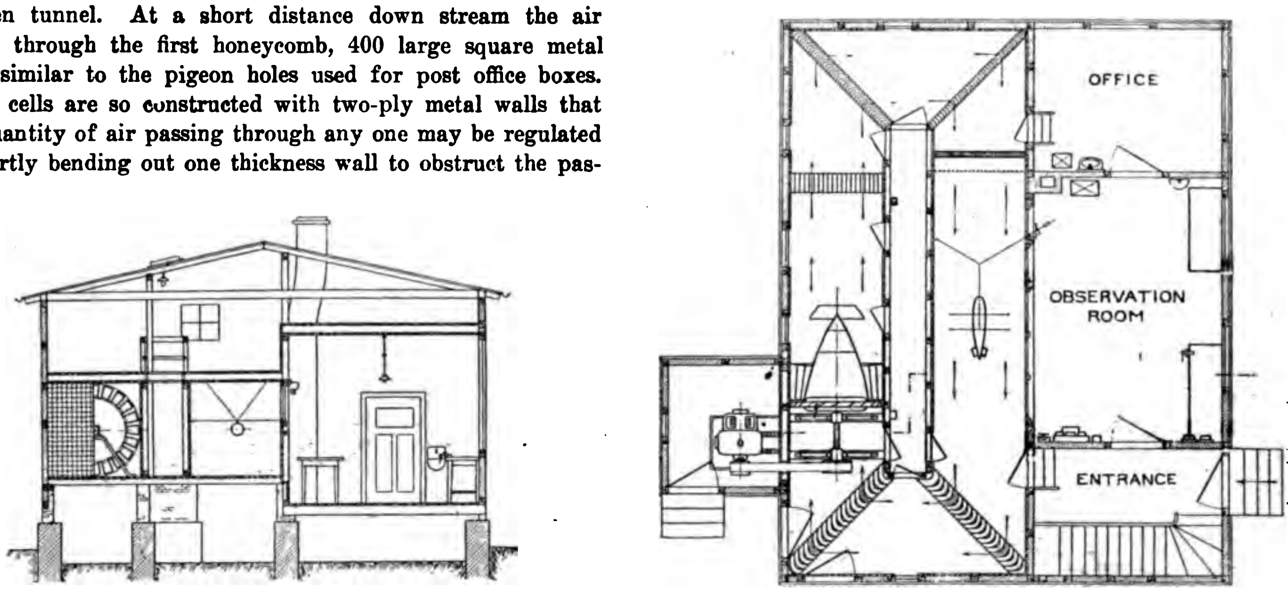


Fig. 4. Longitudinal Section of the Large Wind Tunnel in Eiffel's Laboratory

The Deutsche Versuchsanstalt für Luftfahrt zu Adlershof, superintended by Prof. Dr. Bendemann, is of the same order as the experimental grounds at St. Cyr, but on a much less elaborate scale. The work is principally on full scale airplanes and block tests on aeronautic motors. One building is devoted to full scale testing, another to construction and repairs and five smaller ones to the housing of motor testing apparatus. The main building has a central tower some 100 feet in height from which wind observations may be made and other atmospheric conditions recorded. Cables from the top of this tower are used to support full size airplanes in the determination of their moments of inertia. A track outside of the building is used, as at St. Cyr, for the testing of full size airplanes or surfaces. In this instance a locomotive used to push the mounting stage is substituted for the St. Cyr electric driven car, a rather doubtful adjunct.

The Göttingen Aerodynamical Laboratory, under the supervision of Professor Prandtl, has little of the ornate as compared to the Eiffel Institution, housed as it is in a plain one story brick building, 30 by 40 feet in size. The building, as may be seen from the drawing, is about equally divided between wind tunnel and office space. Glass doors in the side next the observation room permit of access to the experimental section of the tunnel, while trap doors open here and there to allow entrance into other sections for the adjustment of the honey-combs, baffles, etc.

Unlike the Eiffel tunnel, the air follows a closed circuit necessitating the turning of four corners. The 2 meter diameter blower, driven by a 30 h.p. electric motor, forces a steady current of air through the 2 meter, 6.6 foot, square wooden tunnel. At a short distance down stream the air passes through the first honeycomb, 400 large square metal cells, similar to the pigeon holes used for post office boxes. These cells are so constructed with two-ply metal walls that the quantity of air passing through any one may be regulated by partly bending out one thickness wall to obstruct the passes through the second honeycomb, much finer than the first. This last honeycomb is constituted of about 9,000 cells from which the air, after passing a wire mesh to remove any foreign matter, issues with a maximum velocity of 10 meters or 32.8 feet per second, to act upon the model suspended some distance down stream.



Section Plan

FIG. 5. THE GÖTTINGEN AERODYNAMICAL LABORATORIES

sage. The cells, in many instances, have been so restricted to regulate the air flow that it might be as uniform as possible. Vanes, similar to those of a turbine, are utilized at the four corners to turn the current through a 90 degree angle, without producing excess eddy-motion. After the second turn, just before the air enters the experimental part of the tunnel, it

A great deal of the work in the Göttingen Laboratory has been devoted to the resistance of airship hulls, etc., for which work a special suspension method of mooring wires, bell cranks and weights, has been adopted with great success. A differential pressure gage, sensitive to pressure changes of one-millionth of an atmosphere is used in the determination of

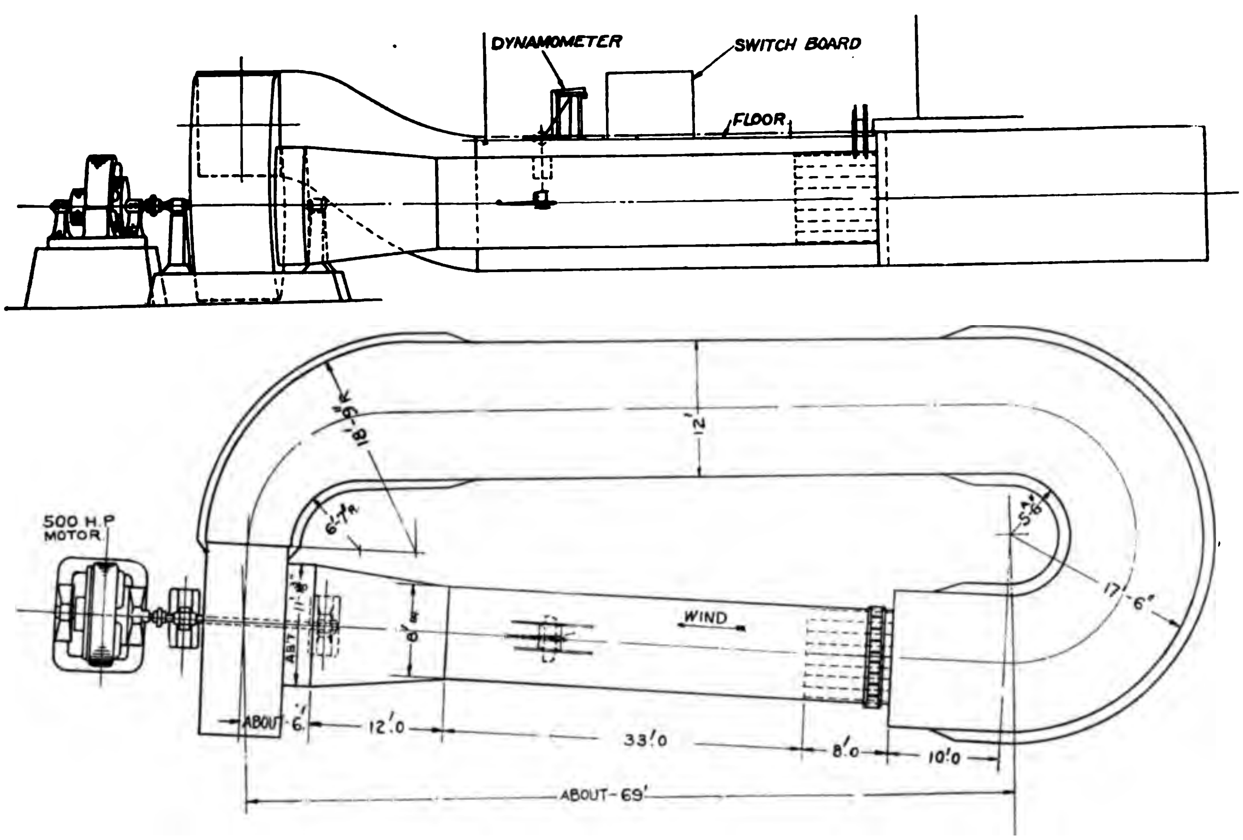


FIG. 6. ELEVATION AND PLAN OF THE WIND TUNNEL OF THE UNITED STATES NAVY DEPARTMENT

velocity. Many interesting experiments on the distribution of pressure have been conducted upon small propellers, constructed by electroplating with copper, wax models. A more detailed description of the suspension device and differential gage will follow.

The Wind-tunnel of the United States Navy Department, under Naval Constructor Holden C. Richardson, is at the Washington Navy Yard, Washington, D. C. The tunnel is similar to the German Göttingen Laboratory in that the air is confined in a closed circuit, in this case eight feet square at the test section. The cross sectional dimensions vary as may be seen in the accompanying print, in order to compensate for the curves taken by the stream. Only one set of honeycomb baffles is employed, these being placed just at the entrance of the experimental chamber and 20 feet up stream. These 64 cells,

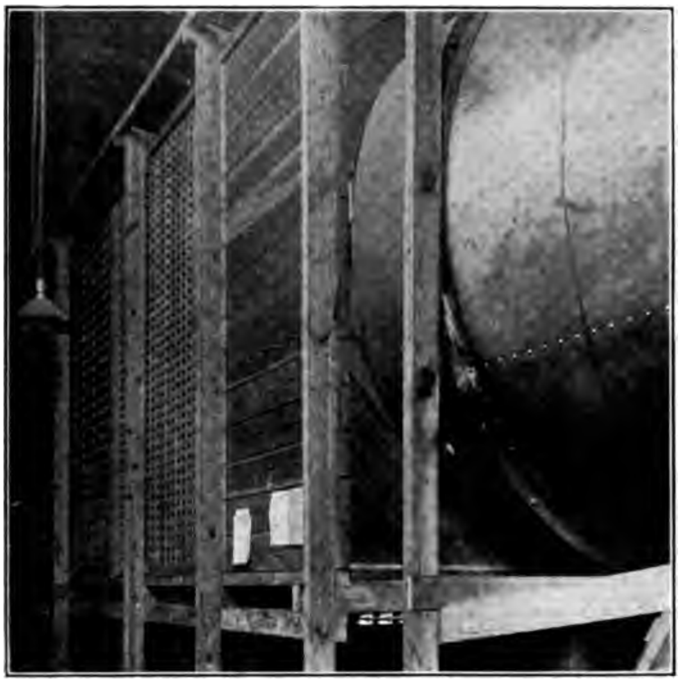


FIG. 7. EXTERIOR OF THE WIND TUNNEL AT THE MASSACHUSETTS INSTITUTE OF TECHNOLOGY, SHOWING CELLS THROUGH WHICH AIR IS SUCKED IN FROM THE ROOM

each one foot square and eight feet long, are equipped with individual adjustable dampers used as a control upon the quantity of air passing and so producing uniform flow to within about 2 per cent. The balance and motor control are mounted on a platform upon the roof of the tunnel. The model is supported in a horizontal position in the wind on a balance similar to that of Eiffel's and sensitive to at least 2/1000th of a pound. Models up to 36-inch span are permitted without noticeable interference from the walls or choking of the air flow.

The velocity measurements are unique in that in place of a single pitot and pressure tube, placed in the vicinity of the model, a series of twelve tubes equally spaced, directly on the discharge side of the blower record on an integrating manometer the velocity of the stream. The velocity of the stream from the blower has a direct relation to the velocity of the wind in the experimental chamber, against which it has been calibrated for all speeds. The pitot tubes used have been themselves checked with the standard tubes of the National Physical Laboratory of England and the Aerodynamical Laboratory of the Massachusetts Institute of Technology.

Power for driving the suction blower is supplied by a 500 h.p. 250 volt direct current electric motor, operated on the Ward-Leonard system. A velocity of 75 miles an hour may be obtainable, but due to the heating of the air by friction and other difficulties in maintaining regular flow this high speed is seldom utilized. Generally tests are made at a speed of about 40 miles an hour.

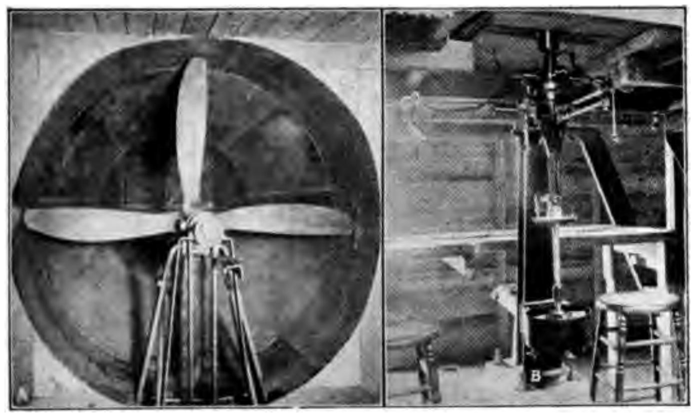


FIG. 8. (A) PROPELLER AND (B) AERODYNAMIC BALANCE IN USE AT THE MASSACHUSETTS INSTITUTE OF TECHNOLOGY

The National Physical Laboratory at Teddington and the Royal Aircraft Factory at Farnborough, England, constitute the most complete aeronautical experimental combination in the world. The aeronautical portion of the National Physical Laboratory is devoted to experimental investigations of the British Advisory Committee for Aeronautics. This committee, with Dr. R. T. Glazebrook as chairman, and such able co-

FIG. 9. ENTRANCE NOZZLE SHOWING HONEYCOMB

workers as Dr. Stanton and Mr. L. Bairstow, initiates the investigations at the N. P. L. and oversees the general work in aeronautics throughout the Kingdom.

The Royal Aircraft Factory, superintended by Mervyn O'Gorman, works in close co-operation with the N. P. L. It has facilities for model experiments, but is more concerned with tests on full size airplanes and the application of the investigations of the National Physical Laboratory. There is necessarily some overlapping in the work carried on at the two institutions, but no interference.

The Royal Aircraft Factory before the war, was the largest factory then in existence devoted to the manufacture of airplanes. All the experiments are carried on in the large flying field in connection with the factory. Machines equipped with intricate recording instruments are flown under their own power and such important information as: power utilized, angles of pitch, roll and yaw, speed through the air, altitude and control movements are simultaneously recorded. This, in a true sense, is full scale experimental work and the results have been to disclose defects and encourage the improvement and safety of the machines. By the careful application of the model experimental work of the N. P. L. an inherently stable biplane with a speed range of 40 to 80 miles an hour had been produced by the R. A. F. before the war. Improved machines of this type have been of the greatest value to the Royal Flying Corps.

The National Physical Laboratory has turned over ample space for the exclusive use of the Aeronautic Committee, comprising a large and small wind tunnel house, a whirling table house and ample space for any independent investigations. The small and large wind tunnels are of similar character, one 4 and the other 7 feet square in cross-section. Each is mounted in a separate building, the smaller being included in the engineering laboratory building. For details of the small tunnel, reference is made to the description of its duplicate in the Massachusetts Institute of Technology Laboratory. The new 7 foot tunnel only differs from the 4 foot in its dimensions and power. It is 80 feet in length with an air flow of 60 feet a second produced by a low pitched four bladed propeller driven by a 30 horse power electric motor.

A great amount of time was spent in experimenting with this form of tunnel before the committee was satisfied with the results. They have the deep satisfaction of knowing that the artificial wind produced by it, is the most uniform in the world and adaptable to the most scientific research. The current is uniform in velocity, both in time and space, to within one-half of one per cent. The velocity measurements and aerodynamical balance will be described in detail later. It suffices here to say that they are as carefully worked out and results obtained as gratifying as the wind tunnel itself. The work of the committee has been extremely broad and the results are of untold value to aeronautics. The whirling-arm and small water channel, the former used in the calibration of velocity instruments, the latter in the study of stream line flow, are both examples of high engineering skill.

The Wind-tunnel of the Massachusetts Institute of Technology was built after a careful study of European Laboratories, on plans furnished through the courtesy of the National Physical Laboratory. Maintained in connection with the graduate course in aeronautics at the Institute, with the helpful cooperation of Professor Peabody of the Naval Architectural Department, and under the former directorship of Lieutenant J. C. Hunsaker, U. S. N., the work has been of the most commendable character.

The tunnel is housed in a temporary building on the new Technology site in Cambridge, with offices in the main Institute building. Enclosed in a 20x25x66 foot shed, the tunnel is suspended in the center of the room, 6 feet from the floor, so that ample space is provided for the free circulation of air. The illustrations indicate the general form of the tunnel which has an overall length of about 56 feet and a working section 4 feet square. The air which is drawn from the room around the cowled entrance end passes through a honeycomb formed from 3-inch metal conduit pipes 2 feet 6 inches in length into the experimental chamber. This honeycomb helps to straighten out the flow and prevent eddies in the wind.

The experimental chamber reaches from this honeycomb to the expanding trunk, but only the section midway between these points is utilized. The air after passing the model goes through a series of diagonal vanes and enters the expanding trunk. Here the velocity decreases with an increase of static pressure. The expansion in 11 feet of length is to a cylinder of 7 feet diameter. This cone expansion, in the English tunnel is only 6 feet on the 11 foot length. By expansion the pressure difference maintained by the four-bladed propeller is reduced and some turbulence in the wake avoided. The discharge from the propeller is received by a large perforated diffuser with the end opposite the propeller a blank wall. The function of this diffuser is to distribute the air into the room at a uniform rate and at a very low velocity. This is indeed accomplished for the area of the perforation in the diffuser is several times that of the tunnel and when a velocity of 30 miles an hour is maintained in the tunnel, the discharge from the diffuser is hardly noticeable.

FIG. 10. INTERIOR OF DIFFUSER LOOKING FROM PROPELLER

A four bladed black walnut propeller of low pitch, revolving 600 r.p.m. will produce a wind of 25 miles per hour. The propeller is driven by a 10 h.p. electric motor through a "silent" chain. The motor is mounted on a separate concrete foundation, as are likewise the aerodynamic balance and the tunnel proper, to avoid any variation in alignment caused by vibration. The sectional area of the tunnel permits of models of 18 inch span and as an extreme 24 inch span, to be tested at speeds from 6 to 40 miles per hour.

The control of the wind is by sentitive rheostats in the motor field and wind speeds may be kept constant as in the English tunnel to within one-half of one per cent. Measurement of velocity is by means of a calibrated "side plate" in the wall of the tunnel, recording on an alcohol manometer the difference of the pressure in the room and in the experimental part of the tunnel. These manometer readings were calibrated against a standard N. P. L. pitot tube to ascertain the true velocity.

The aerodynamical balance was constructed by the Cambridge Scientific Instrument Company, England, and is a counter-part of the English installation. Most model adjustments are possible from the outside without stopping the wind, thus greatly facilitating the experimental work. The balance is arranged so that complete data for the calculation of the stability coefficients for airplanes is obtainable.

While the laboratory is primarily for research work, investigations are conducted for private individuals or manufacturers at nominal charges.

It is interesting to note that the Curtiss Company has recently erected at Buffalo, under the direction of Dr. A. F. Zahm a wind tunnel similar in construction to that of the N. P. L. and the Massachusetts Institute of Technology.

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Chapter II
Elements of Aerodynamical Theory

Liquid, Fluid, and Perfect Fluid

Both liquids and fluids may be defined as substances which flow or are capable of flowing. A liquid is incompressible and therefore of constant density, a fluid is compressible and of varying density. Thus water is commonly spoken of as a liquid, air as a fluid, yet the hard and fast distinction is unfair, since water itself is slightly compressible.

In the transportation speeds employed in aeronautics, the variations in pressure of the air, and the consequent variations in density are so slight, that the air may also be regarded as incompressible. Thus for a dirigible at a speed of 100 miles per hour the increase in pressure at the nose is only about one per cent. It is only at the tips of fast moving propeller blades that the compressibility of air assumes any importance.

The motion of fluids is so complex that no complete mathematical theory has yet been evolved for it. In hydrodynamics the mathematicians have stipulated a perfect fluid possessing no viscosity. In such a fluid all bodies may move without encountering resistance. Although the conception of a perfect fluid may seem of no practical importance, yet hydrodynamical theory serves as a guide in the theory of aeronautics and we shall have to make occasional reference to this idea.

Density of Air

In setting forth data from the laboratories the air will be assumed as having a temperature of 15° C. and a density of .07608 lbs. per cubic foot at sea level.

Variation of Density of Air with Height

Height (ft.)	*Density (lbs. per cu. ft.)*
0	.0761
500	.0748
1,000	.0734
2,000	.0707
5,000	.0632
10,000	.0523
20,000	.0357

Principle of Relative Motion

We shall assume throughout without further reference that the same resistances will be brought into action whether a body is moving through a fluid or a fluid is streaming past a body, provided the relative motion is the same.

This is an idea which often presents difficulties and is very difficult of theoretical demonstration, yet it is merely a matter of common sense. In *La Technique Aéronautique* of May 15th, 1913, M. Lecornu has given a very sound discussion of this point. We will venture a rough illustration. Imagine a boat propelled through a river at rest at a speed of 5 miles per hour. The oars will exert a certain force of propulsion. Now if the river has a contrary current of 5 miles an hour, the boat will remain at rest relative to the banks, yet exactly the same force will be exercised by the oarsman. There is really nothing more to be grasped underlying the principle of relative motion.

Bernouilli's Theorem for Fluid Motion

In the steady flow of a fluid the current at any point is always in the same direction and magnitude and may be represented by a series of stream lines, or by tubes of flow.

The energy of a fluid consists of three parts: (1) The potential energy, or the energy due to its position of height through which it may fall, (2) The pressure energy, (3) The kinetic energy due to its motion, neglecting the effects of viscosity or friction. Bernouilli's theorem states that along any stream line, the sum of these energies is a constant, and if

g = acceleration due to gravity
h = height
p = pressure
V = velocity
ρ = density*

$$h + \frac{p}{\rho} + \frac{V^2}{2g} = \text{constant}$$

In considering air flow in aeronautics where we deal with a fluid ocean of immense depth, the variations in height are negligible, and the theorem becomes:—

$$\frac{p}{\rho} + \frac{V^2}{2g} = \text{constant}$$

The theorem is of fundamental importance in aeronautics; its proof will be found in any text-book on hydrostatics.

This equation may also be written in the following useful form, by multiplying both sides of the equation by ρ:

$$p + \frac{\rho V^2}{2g} = \text{constant}$$

Total Energy of a Fluid Applied to the Theory of the Pitot Tube

The Pitot tube, so frequently employed in aeronautics to measure the speed of a machine in actual flight, furnishes an excellent illustration of the principles just set forth. In Fig. 1 is given a diagram of such a tube.

Its main function is to measure the velocity of flow for a

* (ρ) is used for Density to prevent confusion with D for Drag and to conform with standard usage.

steady irrotational flow of air, and it is unsuitable for measuring the velocity of turbulent flow, such as that occurring in the vicinity of a fan to give an example.

In practice the Pitot tube is finely rounded so as to give the least possible disturbance to the air flow. It consists of two concentric tubes. The inner one is open to the wind, the outer tube is closed to the wind and is only connected to the surrounding air by a series of fine holes. The tubes are connected to the two arms of a pressure gauge as shown in the figure, and the gauge measures the difference in pressure between them.

The inner tube, open to the wind, brings the air impinging on it to rest, and the pressure in it is therefore a measure of both the static pressure in the stream and of the kinetic energy head of the stream. If p is the static pressure of the stream, V the velocity, the total pressure will be given by

$$p + \frac{\rho V^2}{2g}$$

The outer tube, on the other hand, being closed to the wind, will, if the holes are small enough to prevent velocity having

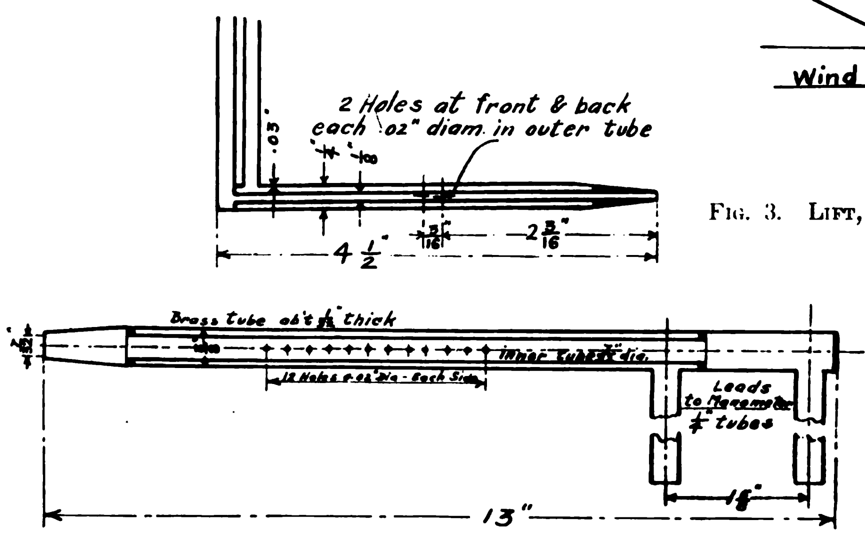


FIG. 1. PITOT TUBE.

any effect on its pressure, read the static pressure of the air flow.

Hence the difference in pressures read on gauge will be

$$\frac{\rho V^2}{2g}$$

and will therefore be a measure of the velocity.

We shall discuss the methods employed in connection with the Pitot gauge more fully when dealing with laboratory methods, but may state now the results of recent experiments as summarized by Dr. J. C. Hunsaker:—

1. The precision is one-tenth of one per cent.
2. The open tube correctly transmits the total pressure regardless of size or shape.
3. The nose of the combination tube must be of easy form.
4. The static openings should be clean holes from 0.01 to 0.04 inches diameter.
5. Static openings should be well back from the nose of the instrument on a polished cylindrical portion of the tube.
6. Static openings may be from 4 to 24 in number arranged in an arbitrary manner.
7. The tube should be pointed into the wind, but an error of two degrees in alignment will cause less than .1 per cent error on velocity measurements.

Definition of Angle of Incidence, Resultant Pressure, Lift, Drag and Center of Pressure in a Plane or Cambered Wing Section

Whether for a plane or a curved wing section, the angle

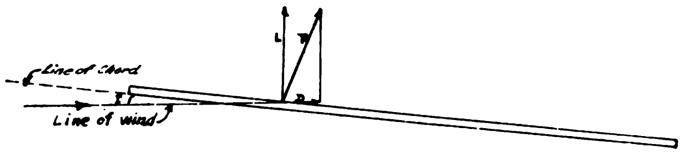


FIG. 2. LIFT, DRAG, ANGLE OF INCIDENCE, AND CHORD FOR FLAT PLATE.

of incidence is defined as the angle i expressed in degrees, between the relative wind and a line in the supporting surface, termed the chord. In the case of the flat plate, this line coincides with the face of the plate and is physically justifiable since when the face of the plate coincides with the relative wind, there is no sustaining force or lift on the plate.

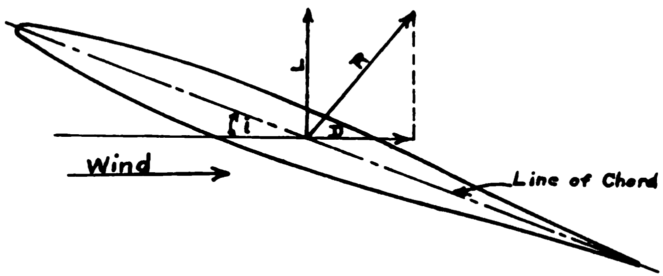


FIG. 3. LIFT, DRAG, LINE OF CHORD, ANGLE OF INCIDENCE FOR DOUBLE CAMBERED SECTION.

In the case of cambered surfaces, the position of the chord has been fixed by conventional usage, and is best illustrated by the diagrams in Figs. 2, 3 and 4. With cambered surfaces, when the conventional chord coincides with the relative wind there is lift as a rule, although the position of no lift may be only a degree or so removed.

Owing to the relative motion of the air, the wing experiences a resultant pressure which we will designate as R. This resultant is very nearly normal to the face of a flat plate, but it is quite wrong to state that it is exactly at right angles to this face. The resultant force R may be generally resolved into two components; one at right angles to the relative wind, which is termed Resistance or Drag (D). Drag will be used instead of the term drift, which unfortunately is capable of misinterpretation. The component at right angles to the relative wind, L, may act upwards, giving *Positive Lift*, or downwards

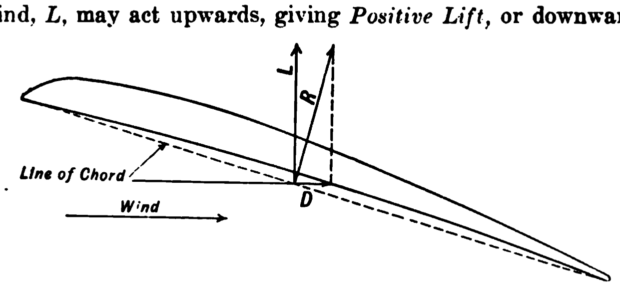


FIG. 4. LIFT, DRAG, ANGLE OF INCIDENCE AND CHORD FOR CAMBERED SURFACE.

giving *Negative Lift*, depending on the position of the surface relative to the wind.

The lift measures the sustaining power, the drag the resistance to forward motion. The tangent of the angle between the R and D gives the ratio L/D, lift over drag. The greater the value of L/D the greater is the path efficiency of the supporting surface.

The center of pressure will be arbitrarily defined as the point of application of the resultant force R on the plane of the wing chord. This is by no means a rigid definition.

Definition of Lift and Drag Coefficients

We shall employ throughout the following notation:

Lift $= L = K_y A V^2$. Resistance or Drag $= D = K_x A V^2$.

Where L and D are in pounds, A = area in square feet of one surface projected on the line of chord, and V = velocity in feet per second, K_y and K_x will represent forces for unit areas and unit velocity. We shall see later the justification for these expressions.

Position of Center of Pressure or Resultant Vector of Forces

It has become customary in Aerodynamics to speak of Centers of Pressure, and it is very often convenient to employ this term. But it would be much better to speak of the position of the resultant vector of forces, a vector being a line representing a force in magnitude and direction. For a flat plate or a cambered wing section, the term center of pressure might answer fairly well, but for a combination of wing surfaces, as in a biplane, or for any kind of airplane, it is very unsatisfactory. Thus as in Fig. 5 for certain angles the resultant force passes right outside the wing surface, and to speak of a center of pressure in such a case is meaningless.

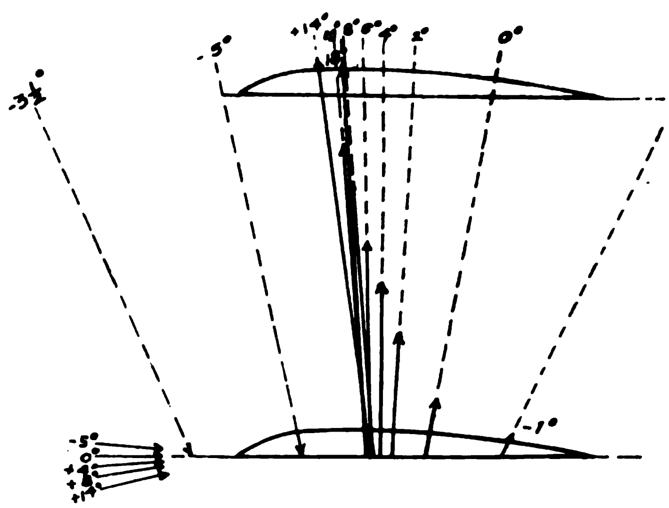


FIG. 5. ILLUSTRATING POSITION OF VECTOR OF RESULTANT FORCES.

It is also often stated that the stability of a wing depends on the motion of the center of pressure with reference to the center of gravity. The moment about the center of gravity can be more correctly stated as depending on the position and direction of the resultant vector of forces. If current practice leads us to speak of center of pressure, the reader will always bear these considerations in mind.

Forces on a Flat Plate Immersed in a Fluid and Normal to the Direction of Motion

Newton was the first to consider the case of a flat plate placed normal to its direction of motion. He stipulated a medium composed of an infinite number of small particles, having no sensible magnitude but possessing mass, and not interconnected in any way. A plate of area A, moving with a velocity V in a medium of density ρ would meet a quantity of fluid $\rho A V$ and impart to this quantity a velocity V per unit of time.

From the fundamental equation in mechanics:

$$\text{Force} = \frac{(\text{mass acted upon}) \times (\text{velocity imparted})}{\text{Time}}$$

we should derive the equation:

$$R = \frac{(\rho A V)}{g} V = \frac{(\rho A V^2)}{g}$$

Similar reasoning from the Principle of Relative Motion would apply were the plate held at rest, and the fluid impinging on it. The force as derived from actual experiment is considerably less than this.

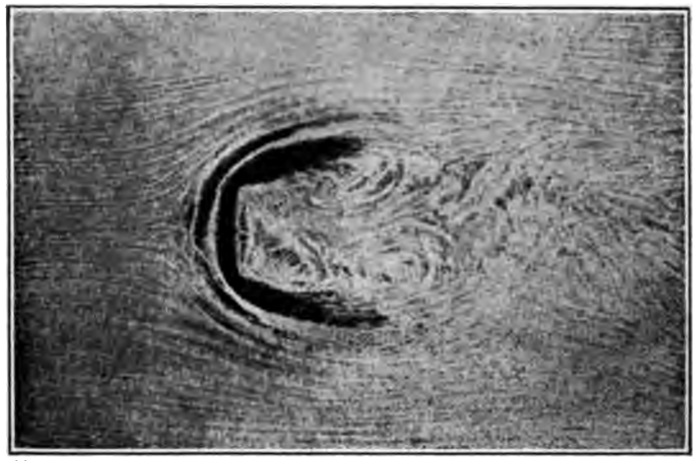


FIG. 6. MOTION NEAR A FLAT PLATE NORMAL TO THE WIND

But Newton's theorem is obviously incorrect, no account being taken of the action at the back of the plate, or of the complicated interaction between the particles, or of the formation of eddies and whirls. The photograph in Fig. 6 gives an idea of the complicated actions which take place. These are represented diagrammatically in Figs. 7 and 8.

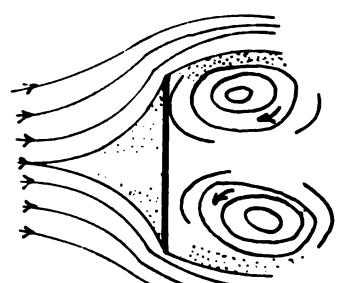


FIG. 7

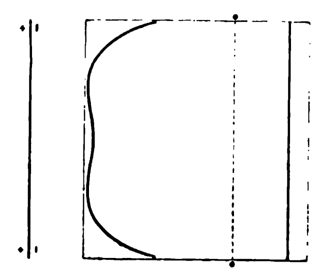


FIG. 8

DIAGRAMS ILLUSTRATING FLUID MOTION AND PRESSURE DISTRIBUTION ON PLATES NORMAL TO THE STREAM

From a consideration of Bernouilli's Theorem it will be seen that the pressure in front of the plate will become greater than the statical pressure of the stream. At the back of the plate, owing to the considerable velocity of the eddies or whirls, we can say again from a consideration of Bernouilli's equations—that the pressure will be less than the statical pressure. It is to the difference in pressures front and back of the plate that the resistance is due. Fig. 8 represents roughly the distribution of pressure on either side of the plate.

Newton was correct, however, in so far as the resultant force of a plate normal to the wind is proportional to the velocity squared, the area, and the density; and if R denotes the resultant force we can write:

$$R = KAV^2$$

where K is an experimental coefficient.

We shall show later that a similar law holds for all cases of bodies producing turbulent flow, and discuss fully the resistances due to such flow.

Forces on Flat Plates Inclined to the Wind

Figs 9 and 10 represent diagrammatically the fluid section in the case of an inclined plate, and the distribution of pressure, which are further illustrated by the photograph (after Riabouchinsky) in Fig. 11.

Just as in the case of the plate normal to the wind, the resultant force will be determined by the difference in pressures at the front and back of the plate, and lift and drag will vary as AV^2, as in the case of all bodies producing turbulent flow, with a different coefficient for each angle of incidence.

The minimum resultant force of a plate occurs when it is in the line of the wind. As the angle of incidence increases so does the pressure, until a critical angle of some 40 degrees is reached. After this, the resultant force slowly diminishes to the value in normal presentation.

At small angles the center of pressure is near the mid position, and gradually moves forward as the angle of incidence increases. That the center of pressure should be forward of the mid position is fairly obvious from the above mentioned photograph. It is in the forward region of the plate that the air experiences the most abrupt changes of direction, with consequently the greatest variation of pressures. This can be seen also from the diagram of distribution of pressures.

Numerous efforts have been made to deduce expressions for lift and drag and for the motion of the center of pressure from theoretical considerations. But the only trustworthy values are those directly taken from experimental data obtained by Eiffel and others which will be dealt with later.

It may be stated here, to remove a somewhat common misconception, that the resultant pressure on a flat plate is not perpendicular to the plate except for a certain limited range of angles of incidence. At zero degree of incidence the resultant pressure is 90 degrees behind the normal, rapidly approaches the normal at small angles, and shoots past it at 10 degrees.

FIG. 9.

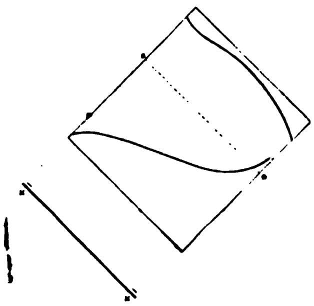


FIG. 10.

DIAGRAMS ILLUSTRATING FLUID MOTION AND PRESSURE DISTRIBUTION ON INCLINED PLANE.

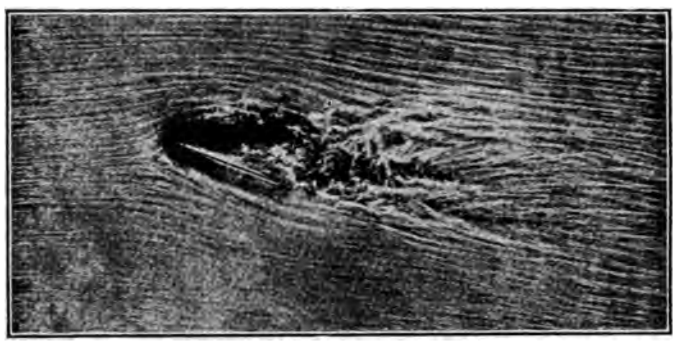


FIG. 11. MOTION NEAR A FLAT PLATE INCLINED TO THE WIND

Chapter III

Elements of Aerodynamical Theory—*Continued*

Skin Friction

Skin friction may be defined as the total resistance of a thin plate moving edgewise through a fluid, and is due to two components:

(1) Viscosity resistance

(2) Density resistance

which we shall consider in turn.

In some respects skin friction is a misleading term. We shall see shortly that the skin of a body has nothing to do with the resistance, a moving body being covered with a layer of fluid at rest. Its usage, however, has been sanctioned by time.

Viscosity

Real fluids like air and water offer a resistance to shear, which is a measure of their viscosity.

Let us imagine two horizontal planes, one of which, AB,

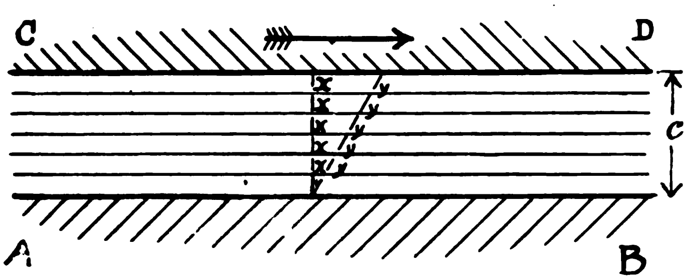


FIG. 12. VISCOSITY ACTION FOR THIN SURFACES.

is at rest, as in Fig. 12, while the other, CD, is dragged past with a velocity V, with the viscous substance intervening, the distance between the two plates being c.

Particles of the substance nearest to CD will adhere to it. Other particles will be carried along to the line *yyyyy* a constantly decreasing amount xy. If F is the horizontal force per unit area required to drag CD, it is obvious that it will be proportional to some constant dependent on the nature of the substance and on the velocity gradient.

We may then write

$$F = \mu \frac{V}{c} \text{ where } \mu \text{ is some constant.}\dagger$$

If V and c are unity

$F = \mu$ and μ becomes the coefficient of viscosity.

The simplest case of viscous drag is that of a thin plate moving edgewise through a fluid. Length is l and breadth b. There will be a thin boundary of fluid of thickness a which connects the particles adhering to the body with the particles at rest in the fluid. This layer will continually lose and gain fluid as it is rubbed off. In unit time a mass of fluid proportional to the cross section of the layer, (ba), and to the velocity, will be captured and have its velocity partly destroyed.

† μ is Greek letter mu.

The inertia force required for this change of momentum will therefore be proportional to

$$(\rho\, ab\, V)\, V \text{ or } \rho\, ab\, V^2$$

The viscous drag must be equal to this inertia force; and is itself proportional to $\mu\ (bl) \times \frac{V}{a}$ by the definition given above. And if

$$\mu\ (bl)\ \frac{V}{a} \sim \rho\ ab\ V^2\text{, then, } a \sim \sqrt{\frac{\mu l}{\rho V}}.$$

The viscous drag therefore is proportional to $\mu\ (bl)\ V \sqrt{\frac{\rho V}{\mu l}}$

$$\text{or to } \mu^{.5}\, b\, l^{.5}\, \rho^{.5}\, V^{1.5}$$

Coefficients of Kinematic Viscosity

To represent the relative importance of density and viscosity a coefficient

$$\nu = \frac{\mu}{\rho}$$

is employed, known as the coefficient of kinematic viscosity. Substituting from this equation in the expression for viscous drag we obtain

$$F \sim \nu^{.5}\, bl^{.5}\, \rho V^{1.5}$$

which may be expressed in the more practical form

$$R_v = d\nu^{0.5} A_s^{0.75} V^{1.5}$$

where R_v = viscous drag

d = constant

A_s = area in shear

$A_s^{.75}$ is equivalent to $bl^{.5}$ dimensionally.

Reynold's Number

It is interesting to note that the thickness of the boundary layer

$$a \sim \sqrt{\frac{\mu l}{\rho V}} = \sqrt{\frac{\nu l}{V}}.$$

The expression

$$\frac{l}{a} \sim \sqrt{\frac{Vl}{\nu}} = \sqrt{r}$$

will be of use in comparing resistances for similar bodies in the same fluid.

It is known as Reynold's number and expresses mathematically a relationship between velocity linear dimensions and viscosity. We shall have frequent occasion to refer to it in comparing resistances of stream line bodies, rods, wires and so forth.

Prandtl's Theory of the Boundary Layer

The theory of the thin boundary layer is due to Dr. Prandtl of Göttingen, his hypothesis being that the velocity gradient is at first very steep but flattens out quickly, until in the free stream the velocity gradient between stream lines is negligibly small. Elaborate experiments by Dr. Prandtl bear out this theory, and demonstrate that the viscous drag does indeed vary as $V^{1.5}$.

Dr. Zahm's experiments on skin friction on the other hand have shown that for even surfaces, bodies covered with such widely varying substances as dry varnish, wet varnish, water, sheet, zinc, etc., all experience the same frictional resistances. It seems therefore reasonably safe to assume that viscous drag is due to internal fluid friction and not to the sliding of the fluid along the surface of the solid.

Density Resistance to a Plate Moving Edgewise

For exceedingly small velocities, it has been found that resistance varies as V^1 indicating purely a drag due to shear (Stokes). For small velocities experiments by Allen have shown a resistance varying as $V^{1.5}$ indicating the condition of viscous drag which we have developed in the preceding paragraphs. But for the velocities with which we are concerned the resistance of a thin plane surface moving edgewise increases as some higher power of V. This is probably—although it is impossible to state the exact cause—due to the fact that the viscous drag not only imparts translational velocity to the particles which adhere to it in the boundary layer but the boundary layer acting as a species of gearing also gives some eddying or rolling velocity to particles adjacent to this boundary layer. It is a commencement of turbulent, eddying motion. As such this extra resistance is proportional to some area A of the body, and to the velocity V, squared, and is termed density resistance and

$$R_d = KAV^2$$

where K is a constant for the fluid.

Total Skin Friction. Dr. Zahm's Experiments

$$\text{Total skin friction} = R_t = R_v + R_d$$
$$= \text{viscosity resistance} + \text{density resistance}$$

Strictly speaking if $R_v \sim V^{1.5}$ and $R_d \sim V^2$ no one expression with V raised to a power n can satisfy this expression. But

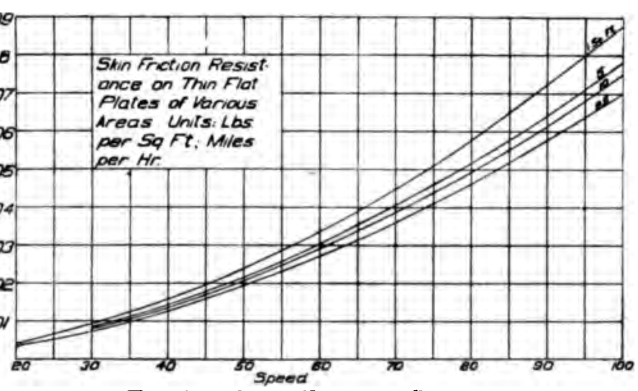


FIG. 13. SKIN FRICTION CHART

for practical purposes the results of Dr. Zahm's valuable experiments have been accepted, his formula being:

$$R = 0.00000778\, l^{.93}\, V^{1.86}\, b$$

where R = resistance for one side of board
l = length in direction of wind in feet
b = width in feet
V = velocity in feet per second.

In the British Technical Report of the Advisory Committee for Aeronautics, 1911-1912, p. 34, an alternative form of equation has been submitted, so as to make the equation consistent with the principles of dynamic similarity:

$$R = 0.0000082\, A_s^{.93}\, V^{1.86}$$

where A_s = area of one side of the board in square feet.

Amongst other applications, Dr. Zahm's formula may be used to compute the resistance of flat rudders, elevators, and stabilizers when neutral to the wind. In Fig. 13 curves for the resistance of plates of various area at varying speeds have been plotted, to facilitate such computations.

Dr. Zahm's skin friction experiments are described in *Bulletin*, Vol. xiv, pages 247-276, of the Philosophical Society of Washington, June, 1904. The plane was suspended in the wind tunnel as shown in sketch in Fig. 14, with wind shields at either end so as to give purely tangential forces.

FIG. 14. ARRANGEMENT OF THIN PLATE IN WIND TUNNEL IN DR. ZAHM'S EXPERIMENTS.

As the wind friction moved the plane edgewise the displacement was determined by the motion of a sharp pointer attached to one suspension wire and traveling over a fine scale lying on the top of the tunnel, and hence the forces were deduced. A variety of shapes and surfaces were tried.

Curves for Computations with Dr. Zahm's Formula

TABLE 1.—SKIN FRICTION RESISTANCE

Speed (feet per second)	Area sq. ft.)							
	1	5	10	15	20	25	30	35
30	.0046	.020	.039	.056	.074	.091	.108	.124
40	.0078	.035	.067	.097	.127	.156	.185	.210
50	.0119	.053	.101	.147	.193	.240	.280	.320
60	.0165	.074	.141	.210	.270	.330	.390	.450
70	.022	.099	.189	.280	.360	.440	.520	.610
80	.028	.127	.240	.350	.460	.570	.670	.780
90	.036	.159	.300	.440	.580	.710	.840	.970
100	.043	.192	.370	.530	.700	.860	1.02	1.18
120	.060	.270	.510	.740	.970	1.19	1.42	1.65

TABLE 2.—ZAHM'S FORMULA

$R = 0.0000167\, A^{0.93}\, V^{1.86}$, V = miles per hour

Speed (miles per hour)	Area (sq. ft.)							
	1	5	10	15	20	25	30	35
30	.0094	.041	.080	.114	.151	.186	.220	.250
40	.0159	.071	.137	.198	.260	.320	.380	.430
50	.024	.108	.210	.300	.390	.490	.570	.650
60	.034	.151	.290	.430	.550	670	.800	.920
70	.045	.200	.390	.570	.730	.900	1.06	1.24
80	.057	.260	.490	.710	.940	1.16	1.37	1.59
90	.073	.320	.610	.900	1.18	1.45	1.71	1.98
100	.188	.390	.750	1.08	1.43	1.75	2.06	2.41

In Fig. 13, the skin friction resistance in lbs. per square foot is plotted against the speed in miles per hour. Since the resistance increases less rapidly than the area, separate curves have been drawn for several different areas, and the force per unit area on any other surface can be found by interpolation. The curves were plotted by modifying Zahm's formula:

$$R_t = 0.0000082\, A^{.93}\, V^{1.86}$$

where V is in feet per second. To throw this into mile hour units it was necessary to multiply by $(\frac{22}{15})^{1.86}$, or 2.04, giving

$$R_t = 0.0000167\, A^{.93}\, V^{1.86}$$

In Tables 1 and 2 similar data has been given for speeds in feet per second and miles per hour.

Turbulent Flow, Eddy or Density Resistance

We have already seen in the case of a flat plate normal to the wind that the resistance was due to a region of turbulent, eddying, low pressure behind the plate. This resistance varies as $\rho A V^2$

where A = area in normal presentation
ρ = density
V = velocity.

It will be assumed for the time being that wherever there is a region of turbulent, eddying flow, there will be a density resistance

$$R \sim \rho A V^2$$

A fairly complete demonstration of this has been given by a French author.

Comparison of Forces Acting Upon Similar Bodies. The Importance of Kinematic Viscosity and the Reynold's Number

For the comparison of forces acting on similar bodies, a knowledge of the geometric proportions and of the wind velocities is insufficient. The density of the fluid, the viscosity and hence the coefficient of kinematic viscosity, and the compressibility of the fluid all enter into the complete comparisons.

Compressibility, we have seen, may fortunately be neglected in all aerodynamical work.

For bodies in which the resistance is purely of a density or eddy making nature—as in the case of a flat plate normal or inclined to the wind, and as we shall see subsequently in the case of a wing section at large angles—viscosity does not enter into consideration, or is of so small importance that it may be neglected. In such cases

$$R \sim \rho A V^2$$

where A is the area of one face of the plate, comparison between two bodies such as a full sized wing and its model become extremely simple.

But for stream line bodies such as struts, cables, wires, and cylinders the resistance is compounded of density resistance and viscosity resistance in varying proportions.

Viscosity resistance depends, as we have seen, on velocity, linear dimensions and the coefficient of kinematic viscosity. For such bodies therefore the resistance must be expressed in a form involving these variables, and by the principle of dynamic similarity it can be demonstrated that

$$R = \rho l^2 V^2 f\left(\frac{lV}{\nu}\right)$$

where f is some unknown function and l^2 is of the same dimensions as A. The $\rho l^2 V^2$ brings out the density resistance, $f\left(\frac{lV}{\nu}\right)$ the viscosity resistance; since $\frac{lV}{\nu}$ = the Reynold's number r, we can write

$$R = \rho l^2 V^2 f(r)$$

The reader will now appreciate the importance of the Reynold's number in comparing the resultant forces on the above mentioned bodies.

It is quite incorrect to compare such bodies, making allowance for variation in l^2 and V^2 only unless the Reynold's number is the same for the two bodies under comparison.

In practice it is very rare that comparisons of forces are made with reference to two different fluids. We are almost solely concerned with bodies in air. The coefficient of viscosity becomes a constant, and instead of considering the Reynold's number, we can drop the ν and compare bodies having the same product lV.

Stream Line Bodies

A stream line body may be defined as one which has a gradual change of curvature along any section, and which when moved through air or water at ordinary speeds makes little disturbance or turbulent wake. Such a body moving in a viscid fluid would experience mostly frictional resistance.

Energy Considerations for a Perfect Fluid Flowing Past a Stream Line Body

It is most useful to have a definite idea of the exchange of energy which occurs in such a case. The first treatment appears to have been given by W. Froude.

Imagine the fluid in the vicinity of the body to be divided up into a large number of imaginary tubes of flow. Well ahead of the body where the stream is as yet undisturbed the energy of the fluid will be that due to the static pressure p_0 of the stream and the kinetic energy head of V_0, the undisturbed velocity. In a perfect fluid this will remain a constant along any tube of flow by Bernouilli's theorem, and is equal to

$$\frac{p_0}{\rho_0} + \frac{V_0^2}{2g} = C = \frac{p}{\rho} + \frac{V^2}{2g}$$

For the portion L, as shown in Fig. 15, of the body, the tubes

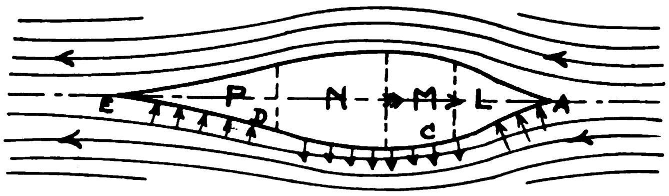


FIG. 15. LINES OF FLOW FOR A STREAM LINE BODY.

of flow widen out, the velocity and the kinetic energy head diminish and the pressure on the body becomes greater than the static pressure p_0. The nose of the body therefore does work upon the fluid in contact with it. This is also evident by considering the effect of curvature and the centrifugal force resulting from it. For the portion M the tubes crowd together, the velocity increases and the body is under the action of a pressure less than p_0—it is really under suction and the fluid does work on the body. By similar reasoning it can be shown that the portion N the body works upon the fluid, and for the portion P, the fluid works upon the body. The balance of work done on the body is thus found to be zero.

Stream Line Bodies in a Viscous Fluid

At slow speeds in water almost perfect stream line motion has been observed and recorded by Dr. Ahlborn (see Fig. 16). But at ordinary speeds, even with stream line forms, there is

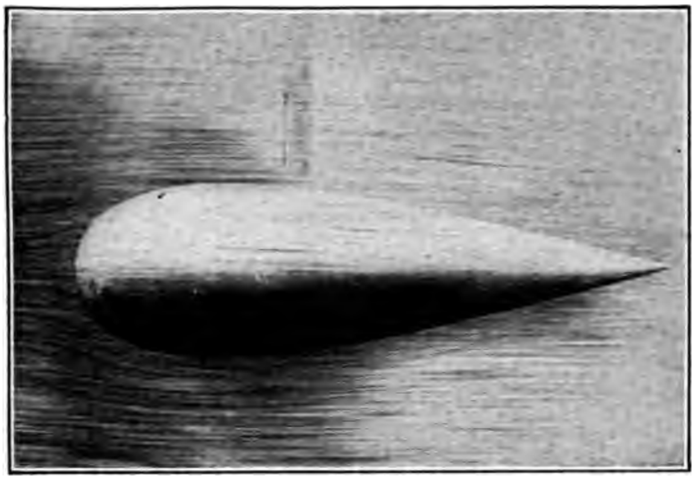


FIG. 16. MOTION AROUND A STREAM LINE BODY.

always a region of turbulence and eddying motion such as we have already observed in the case of the flat plate, accompanied by a surface of discontinuity between the main stream and the turbulent region. The eddying motions are in part due to pressure differences in the undisturbed stream and the region behind the body, in part due to viscosity. The exact theoretical investigations of the causes at play are unimportant from the designer's point of view. It is more important to notice that just as in the case of the flat plate, this turbulent

FIG. 17. FLOW FOR A SHORT STRUT.

region will be a region of low pressure and will introduce a density or eddying resistance.

This density resistance for a stream line body may be said to increase with the extent of the turbulent region. Thus in

FIG. 18. FLOW FOR FINE STRUT.

Figs. 17 and 18 depicting two standard struts, the finer strut has a smaller turbulent region and considerably less resistance. On the other hand, as the fineness ratio, or the ratio of length to maximum thickness, of a stream line body increases, the area in shear and the viscosity drag increase also; the fineness ratio must be kept within reasonable limits even from a purely aerodynamic point of view.

Resistance of Wires, Cables and Cylinders

Fig. 19 represents diagramatically the fluid motion round a cylindrical body, such as a wire or cable, at usual airplane speeds.

It is obvious that the resistance will be partly due to viscosity over the front part of the cylinder, and partly due to eddy or density resistance. The forces in action will therefore be

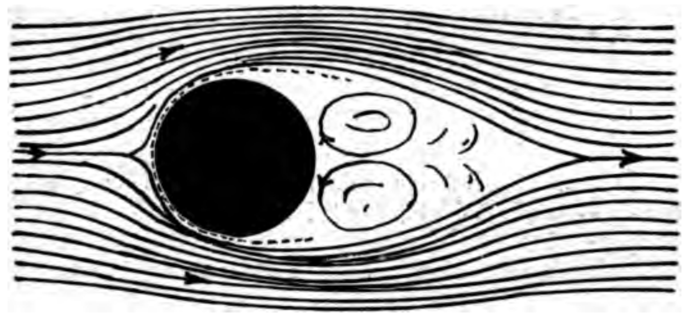


FIG. 19. FLOW AROUND A CYLINDER.

represented as previously stated by an expression of the form

$$\rho l^2 V^2 f(r).$$

And two wires or cables will only be comparable when r is the same for both, or simply when the product lV is the same.

Fluid Motion Around Wing Surface

It is to Langley, above all other men, that we owe an appreciation of the value of cambered surfaces. A good wing sec-

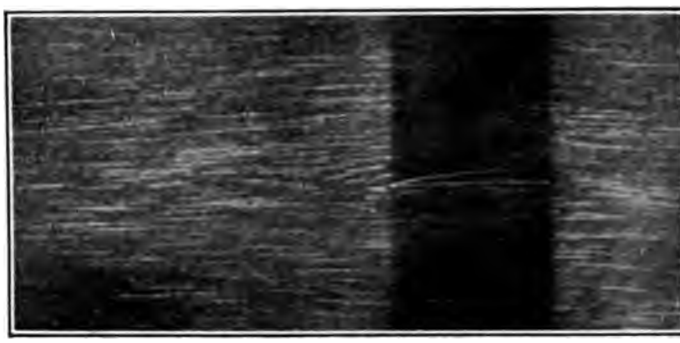


FIG. 20. FLOW FOR A CAMBERED WING AT 2°.

tion may give a lift-drift ratio of 18 as compared to the 6 or 7 of a flat plate, and it is the remarkable efficiency of a wing surface which has largely rendered aviation possible.

In wing surfaces, we recognize two distinct types of flow. For the small angles up to 6° or thereabouts a steady flow as shown in Fig. 20 for a typical airplane wing. At this angle—

FIG. 21. FLOW FOR A CAMBERED WING AT 10°.

often termed the first or lower critical angle, turbulence begins. At 10°, as shown in Fig. 21, this turbulence is quite considerable. Finally a second critical or "burble point" is reached at 18° for the same wing. Here an extremely turbulent type of motion, as shown in Fig. 22 is found, and the lift of a wing attains its maximum. Beyond this "burble point" the motion becomes extremely unsteady and the lift decreases.

The lift of a wing, as experiment shows, varies directly as ρAV^2, with a different coefficient for every angle of incidence.

Where turbulent flow is present this is readily explainable,

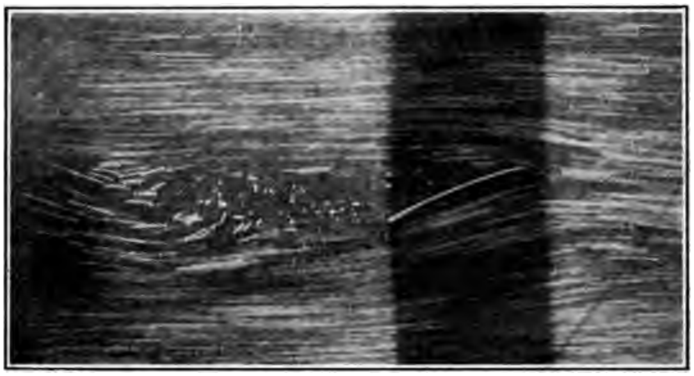


FIG. 22. FLOW FOR A CAMBERED WING AT 18°.

as in the case of the flat plate, on the hypothesis of low pressure region at the back of the wing.

It is the lifting power at small angles and in a condition of steady flow that offers theoretical difficulties. The most likely explanation is offered by Kutta's theory or the vortex theory of sustentation. We shall reserve the full treatment

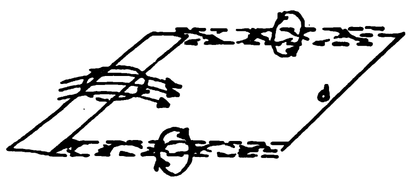


FIG. 23. AEROPLANE WING WITH TRAILING VORTICES.

of this theory also to a special article, contenting ourselves with the barest outlines:

An airplane wing in steady flow gives off a series of trailing vortices as depicted diagramatically in Fig. 23. These vortices are constantly destroyed and renewed. The circular motion in these vortices and their interaction is such that—as the hydrodynamical theory demonstrates—they have a downward momentum, and action and reaction being equal, the airplane wing receives an upward momentum.

The drift or drag of a wing is for all practical purposes taken as varying directly with AV^2, with a different coefficient for every angle of incidence.

At high angles of incidence, the drift is almost entirely a component of the density resistance, and we see that what is taken to be the case in practice, is also theoretically correct. But at small angles and steady flow the resistance is more of a viscous nature, more akin to skin friction. And skin friction, as we have already seen, varies as $V^{1.85}$, and depends also on the dimensions of the body. This introduces considerable difficulties, as we shall see later, in computing resistance in actual flight from small model experiments at low speeds.

As to the form of wing giving the best results, no general laws are yet available, and each type of wing must be considered separately.

This section constitutes but a brief introduction to aerodynamical theory, but will perhaps assist the reader in the appreciation of the extensive aerodynamical data which we shall present later.

References for Chapters 2 and 3

"A Review of Hydrodynamical Theory as Applied to Experimental Aerodynamics," J. C. Hunsaker, International Engineering Congress, San Francisco, Sept., 1915. An authoritative and advanced treatment; containing numerous references for further reading.

"Aerodynamics." F. W. Lanchester (Archibald Constable & Co., Ltd., London.) A valuable, classical treatise.

"The Aeroplane," A. Fage (Griffin & Co., London). A concise scientific study, excellent in matter and presentation.

"The Aeroplane," T. O'B. Hubbard, J. H. Ledeboer, and C. C. Turner (Longmans, Green & Co., London). An elementary text book of the principles of dynamic flight, suitable for beginners.

"Leitfaden der Flugtechnik," S. Huppert (Springer, Berlin).

"Wind Resistance of Some Aeroplane Struts," Booth and Eden. Technical report of the (British) Advisory Committee for Aeronautics, 1911-1912, No. 49 (Wyman & Sons, Ltd., London).

"Investigation by Visual and Photographic Methods of the Flow Past Plates and Models," Eden. British report, 1911-1912, No. 58.

"Photographic Investigation of the Flow Round a Model Aerofoil," Relf. British report, 1912-1913, No. 76.

These papers in the British report contain some beautiful and instructive photographs.

Chapter IV

Flat Plates: Simple Problems on Sustention and Resistance of Wing Surfaces

Coefficients of Resistance for Circular or Square Plates Normal to the Wind. Varying Sizes

Although it would seem that the question of the forces on a flat plate placed normally to the wind would be fundamental in aeronautics, and although it can be shown by the principle of dynamical similarity that similar plates should have the same coefficients no matter what their size, provided that lV remains constant, yet considerable controversy exists as to the variation in the values with the size of plate, and with the velocity of flow. Those who are interested in the controversial aspect of the question are referred to the references at the end of the section. For all practical purposes, the following table may be safely used.

$R = KAV^2$ where R = resistance in pounds.
A = area of plate in square feet.
V = velocity in miles per hour.

TABLE 1.

Side of Square or Diameter of Circular Plate, in Feet.	K.
0.5	.00269
1.0	.00286
2.0	.00314
3.0	.00322
5.0	.00327
10.0	.00327

Coefficients for Rectangular Flat Plates Normal to the Wind. Varying Aspect Ratio

The aspect ratio of a flat plate is the ratio of b to a as shown in the Fig. 1. With increased aspect ratio the resistance coefficient increases. A thorough theoretical discussion of this involves tremendous difficulties, but the increase is probably due to the fact that with increased aspect ratio the air flow is broken up into a greater number of vortices, with a resultant greater turbulence. The following table shows the effect of increased aspect ratio, according to experiments by Eiffel. The coefficient for a square plate of the same area is taken as unity.

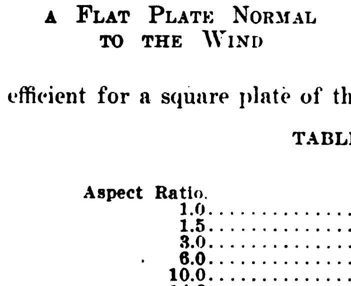


FIG. 1. ASPECT RATIO FOR A FLAT PLATE NORMAL TO THE WIND

TABLE 2.

Aspect Ratio.	K for Rectangular Plate / K for a Square Plate.
1.0	1.0
1.5	1.04
3.0	1.07
6.0	1.10
10.0	1.15
14.6	1.25
20.0	1.34
30.0	1.40
50.0	1.47

These values are plotted in Fig. 2 and are assumed to be true independently of the size of the plate.

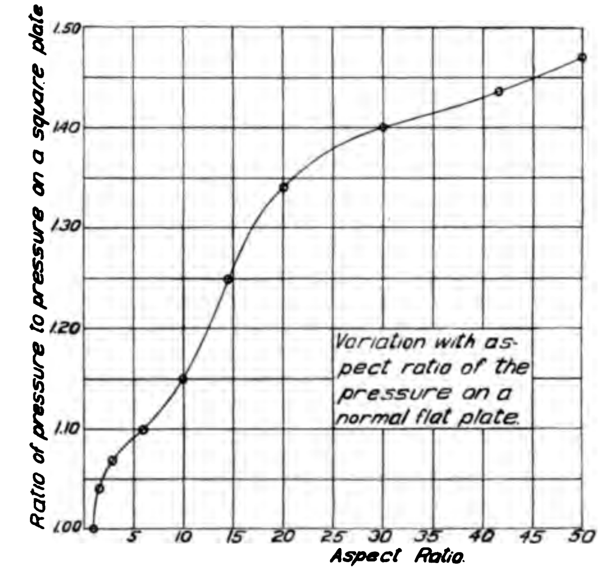


FIG. 2. VARIATION WITH ASPECT RATIO OF THE PRESSURE ON A NORMAL FLAT PLATE

Coefficients for Flat Plates Inclined to the Wind

Table 3 gives values for K_y, K_x, and L/D, and Table 4 the distance of the center of the pressure from the leading edge of the plate in terms of the chord, for flat plates of various aspect ratios. The drag at 0° may be calculated from Zahm's formula for skin friction.

TABLE 3.

Aspect ratio = 1.

Angle.	K_y	K_x	L/D
5	.00045	.00007	6.3
10	.00097	.00019	5.1
20	.00208	.00074	2.8
30	.00294	.00173	1.7
45	.00207	.00210	.99
60	.00139	.00245	.57

Aspect ratio = 3.

Angle.	K_y	K_x	L/D
6	.00090	.00011	7.6
10	.00140	.00028	5.1
20	.00210	.00077	2.7
30	.00193	.00111	1.7
60	.00140	.00236	.59

Aspect ratio = 1.5

Angle.	K_y	K_x	L/D
10	.00109	.00020	5.6
20	.00215	.00077	2.8
30	.00198	.00095	1.7
40	.00183	.00152	1.2
60	.00127	.00226	.57

Aspect ratio = 6.

Angle.	K_y	K_x	L/D
6	.00109	.00018	6.3
10	.00173	.00034	5.2
20	.00199	.00074	2.7
30	.00204	.00120	1.7
60	.00141	.00244	.58

Aspect ratio = 2

Angle.	K_y	K_x	L/D
6	.00074	.00010	7.5
10	.00123	.00021	5.9
20	.00247	.00091	2.7
30	.00178	.00111	1.7
40	.00169	.00146	1.2
60	.00126	.00225	.56

Aspect ratio = 9.

Angle.	K_y	K_x	L/D
6	.00137	.00026	5.2
10	.00186	.00040	4.7
20	.00211	.00080	2.6
30	.00210	.00127	1.7
60	.00140	.00253	.55

Aspect ratio = 1/3.

Angle.	K_y	K_x	L/D
6	.00033	.00007	4.4
10	.00059	.00014	4.2
20	.00138	.00053	2.6
30	.00219	.00132	1.7
45	.00246	.00250	.98
60	.00155	.00264	.58

Aspect ratio = 1/6.

Angle.	K_y	K_x	L/D
6	.00020	.00005	4.0
10	.00040	.00010	4.0
20	.00114	.00045	2.6
30	.00176	.00102	1.7
45	.00217	.00217	.99
60	.00163	.00282	.58

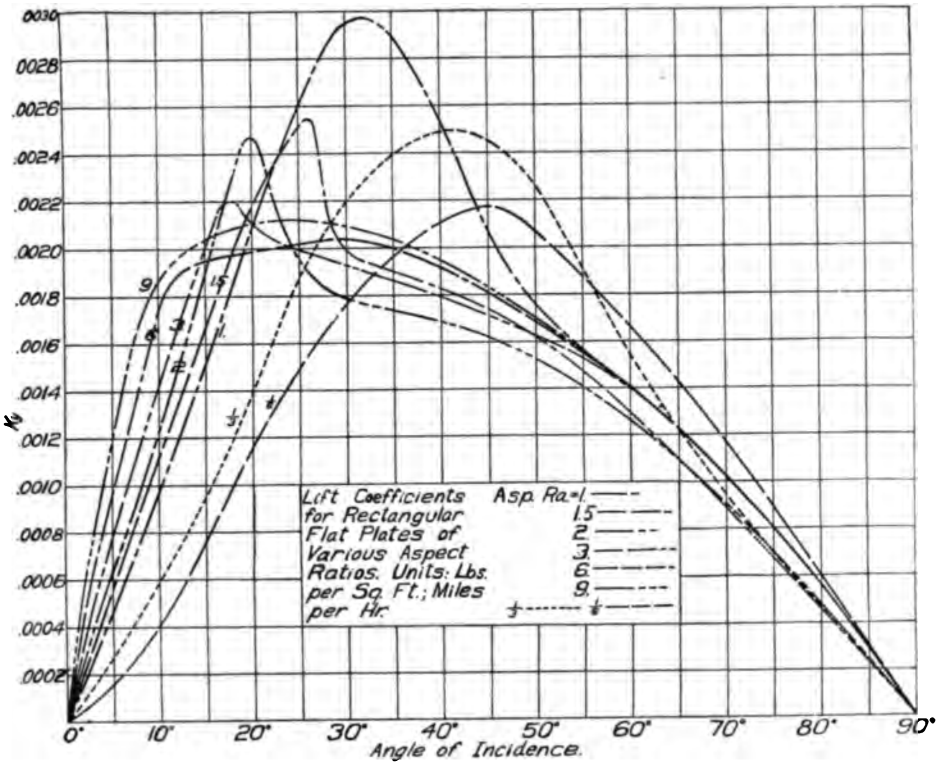


FIG. 3. LIFT COEFFICIENTS FOR RECTANGULAR FLAT PLATES OF VARIOUS ASPECT RATIOS

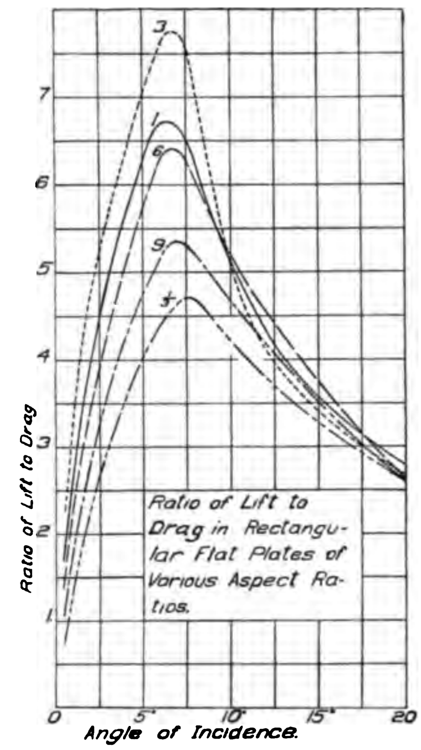


FIG. 4. RATIO OF LIFT TO DRAG IN RECTANGULAR FLAT PLATES OF VARIOUS ASPECT RATIOS

TABLE 4.

DISTANCE OF CENTER OF PRESSURE FROM LEADING EDGE, MEASURED IN TERMS OF CHORD, FOR RECTANGULAR FLAT PLATES OF VARIOUS ASPECT RATIOS.

A. R. = 1.		A. R. = 3.		A. R. = 6.		A. R. = 1/3.		A. R. = 1/6.	
Dist.	Angle.	Dist.	Angle.	Dist.	Angle.	Dist.	Angle.	Dist.	Angle.
.12	.8	.233	5.0	.267	3.0	.167	3.0	.289	2.5
.16	1.0	.267	7.8	.300	8.0	.267	5.0	.311	7.5
.18	2.0	.300	10.0	.333	10.0	.283	6.8	.323	10.5
.20	2.8	.333	12.0	.367	12.2	.300	10.8	.334	19.0
.22	3.8	.367	13.8	.400	26.0	.317	17.5	.345	49.0
.24	6.5	.400	17.5	.433	54.0	.333	30.5	.356	52.0
.28	13.0	.433	52.8	.467	73.7	.350	45.0	.367	53.5
.30	15.3	.467	73.7	.500	90.0	.367	47.8	.378	56.2
.32	18.0	.500	90.0			.383	50.2	.389	58.0
.34	21.0					.400	52.5	.400	59.5
.36	25.0					.417	54.3	.411	60.0
.38	28.0					.433	56.5	.422	63.0
.40	33.5					.450	65.0	.433	64.0
.42	39.0					.467	77.8	.444	68.5
.44	55.2					.483	85.8	.455	72.5
.46	73.5					.500	90.0	.466	80.0
.48	84.0							.477	84.0
								.488	87.5
								.500	90.0

In Fig. 3 are plotted values of K_y against angle of incidence for various aspect ratios. In Fig. 4 the same treatment is applied to the L/D ratio.

In Fig. 5 are indicated the positions of the center of pressure for various aspect ratios and angles of incidence. In Fig. 6 the directions and points of application of the resultant forces are indicated for a flat plate of aspect ratio 6—the value which is usually employed for purposes of comparison—in order to give the reader a more graphic idea of the forces at play.

In all these values it may be noted that no allowance is made for possible variation in the coefficients with size of plates, and this is probably accurate enough for all practical purposes.

Preliminary Application of Data for Flat Plates in Rudder and Elevator Design

These curves and tables give fairly complete data for flat plates and are likely to meet all the requirements of design.

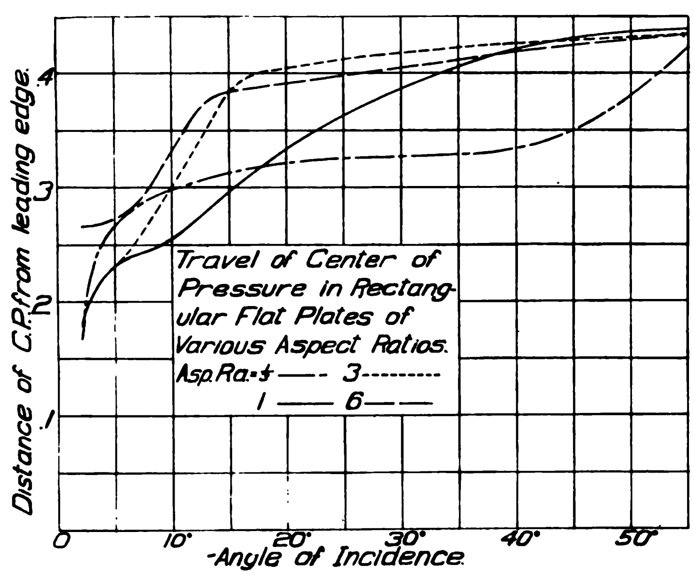


FIG. 5. TRAVEL OF CENTER OF PRESSURE IN RECTANGULAR FLAT PLATES OF VARIOUS ASPECT RATIOS

It may be useful to indicate a few salient points, and to make preliminary reference to the design of flat rudders and elevators.

(1) For plates of all aspect ratios when turned from zero angle, the lift increases until the critical angle or "burble point" is reached. Beyond this angle the lift rapidly decreases, and no rudder or elevator should be employed beyond this critical angle.

(2) The lift drag ratio is not much improved, for flat plates at the same angles, by increased aspect ratio. For all plates the ratio reaches its maximum value at small angles, 6° or 7°. At angles still smaller it decreases, due to the predominating

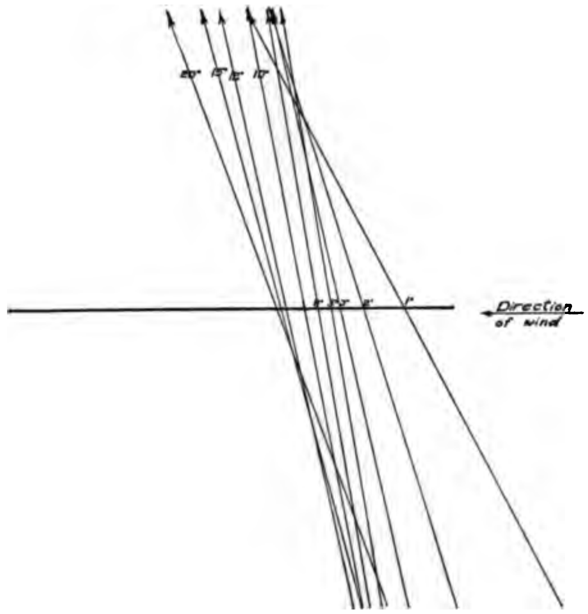


FIG. 6. DIAGRAM SHOWING DIRECTION AND POINT OF APPLICATION OF RESULTANT FORCE IN A RECTANGULAR FLAT PLATE OF ASPECT RATIO 6 AT SMALL ANGLES OF INCIDENCE

effect of the skin friction. Plates of large aspect ratio, being more sensitive at small angles, are, on the whole, more efficient in flight.

(3) On the other hand, plates of small aspect ratio have the critical angle much later and give a wider range of action. They also give a much higher lift at the critical angle, which is important in the action of the rudder when "taxying" at low speeds on the ground.

(4) For the elevator, which is more constantly used in the air, and from which great lifting power is not required on the ground, an aspect ratio of three seems a fair compromise.

(5) For the rudder, the above considerations seem to indicate an aspect ratio of one or two as advisable.

(6) It should be noted that, as the angle of incidence is increased, not only does the force increase, but also that from the point of application of the resultant force to the hinge, giving a greatly increased moment about the hinge. If either the elevator or the rudder is placed too near the wings it

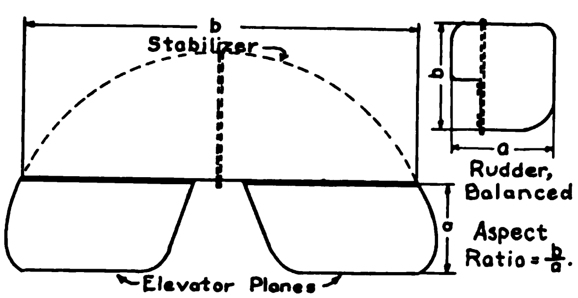


FIG. 7. DIAGRAMS FOR ASPECT RATIO IN RUDDER AND ELEVATOR

necessitates large areas for the controlling surfaces, and the pilot may have to exert tremendous force at large angles.

(7) To obviate the necessity of exercising large forces on the controls, it is possible to use a balanced rudder; one in which the hinge is placed about in the position of the center of pressure at small angles. The rudder in Fig. 7 is a balanced rudder. It should be noted that the "balance" is only approximate.

Problems on Flat Plates

A rectangular flat plate 4 feet 9 inches high and 3 feet 2 inches long is employed as a rudder, and is placed with its leading edge at a distance of 18 feet from the center of gravity of the machine. The machine is traveling at 60 miles an hour. The rudder is hinged at the leading edge, while the control leads are one foot from the rudder surface. (See Fig. 8.) Find (a) the frictional resistance of the rudder when

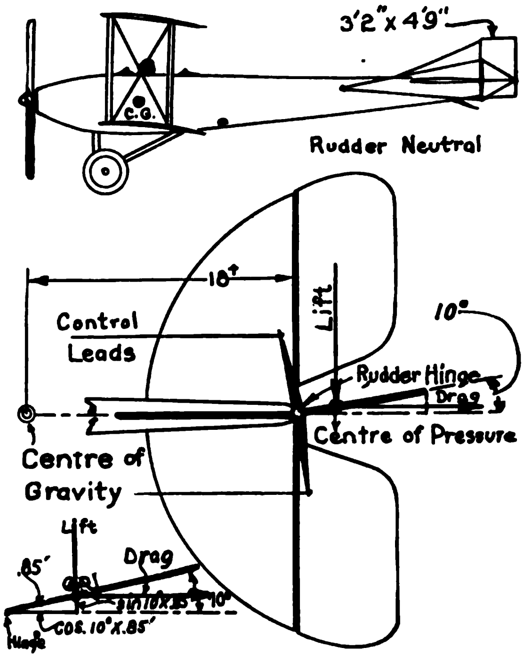


FIG. 8. RUDDER FOR PROBLEM ON FLAT PLATES. RUDDER IS UNBALANCED AND HINGED AT LEADING EDGE

neutral; (b) its turning moment about the center of gravity when set at an angle of 10° and its resistance at that angle; (c) the tension in the control lead under the same conditions as (b).

(a) The area of the rudder $= 2 \times 3\frac{1}{6} \times 4\frac{3}{4} = 30$ square feet. From Fig. 13 in Chapter 3, we see that the frictional resistance on a surface of 15 square feet at 60 miles an hour equals .0285 pounds per square foot. Thus the total frictional resistance $= 2 \times 15 \times .0285 = .862$ pounds.

(b) The aspect ratio of the rudder = 1.5. The distance from the leading edge to the center of pressure is given by Fig. 5. Interpolating between (A.R. = 1) and (A.R. = 3), we see that the center of pressure on a plate of aspect ratio 1.5 at an angle of incidence of 10 deg. is .268 of the chord from the leading edge. Thus the desired distance $= 3\frac{1}{6} \times .268 = .85$ ft.

The moment arm about the center of gravity longitudinally (see sketch of machine) $= 18 + .85 \cos 10° = 18.84$ feet.

The moment arm about the center of gravity laterally $= .85 \sin 10° = .14$ feet.

By Figs. 3 and 4, $K_y = .00109$ and $L/D = 5.5$. Then L, the force perpendicular to the line of flight, $= K_y A V^2 = .00109 \times 15 \times (60)^2 = 58.9$ pounds, and D, the resistance, $= L \times D/L = \frac{58.9}{5.5} = 10.8$ pounds. It will be seen that turning the rudder causes a decided increase in the resistance of the machine.

The above work gives us a basis for rapidly computing the turning moment. $M = 58.9 \times 18.84 + 10.8 \times .14 = 1112$ pound feet, taking the movements of both the lift and the drag about the center of gravity.

(c) The turning moment about the leading edge of the rudder $= 58.9 \times .85 \cos 10° + 10.8 \times .85 \sin 10° = 51$ pound feet.

Moment arm of control lead $= \cos 10° = .986$ foot.

Then, since the stress in the control lead times the moment arm must just balance the turning moment of the rudder about its axis, tension in lead $= \frac{51}{.986} = 52.7$ pounds

General Considerations of Sustaining Power and Resistance of Wing Sections

We have seen that the equation for lift is

$$L = K_y A V^2 \qquad (1)$$

where K_y is a constant varying with the angle of incidence, A = area in square feet, and V = speed in miles per hour.

In horizontal flight, the lift equals the weight of the machine, W, and the equation becomes

$$W = K_y A V^2 \qquad (2)$$

which can be expressed in the forms

$$K_y = \frac{W}{AV^2} \qquad (3)$$

$$V = \sqrt{\frac{W}{K_y A}} \qquad (4)$$

$$A = \frac{W}{K_y V^2} \qquad (5)$$

as may be convenient. The lift coefficient is small at small angles and increases at larger angles until the "burble" point

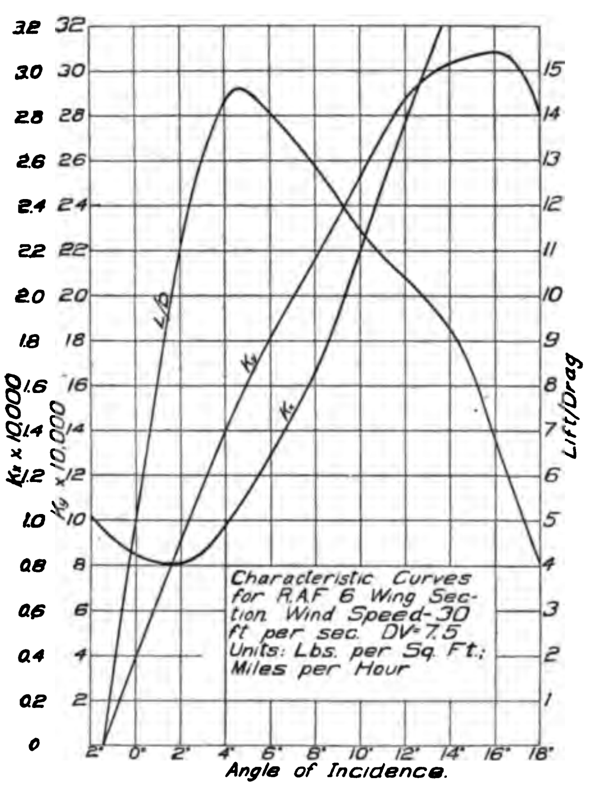


FIG. 9. CHARACTERISTIC CURVES FOR R. A. F. 6 WING SECTION

or critical angle is reached, as can be seen from the curve of a standard wing section (R. A. F. 6) in Fig. 9.

From these considerations may be deduced the following ideas, which should become absolutely familiar to every student of aeronautics:

A machine traveling fast will require, by equation (3), a small value of K_y, and hence a small angle of incidence. Conversely, flying slowly it will require a large angle of incidence.

Sustaining a given weight, we can vary angle of incidence and either area or speed.

If we give a machine a large wing area, it will fly slowly. With a small area, it will attain a high velocity if sufficient engine power is available.

The drag equation is

$$D = K_x A V^2 \qquad (6)$$

The higher the value of L/D, the smaller will be the drag for a given lift and weight of machine at a given speed, and the less will be the power required. The ratio L/D is therefore a measure of the wing efficiency. For the R. A. F. 6, the maximum value of L/D is at about 4° and at about this angle a machine would fly at its greatest efficiency.

We have here neglected all other resistances than those of the wings. These resistances will modify the drag equation and the best angle of flight. We shall deal with these modifications under the Economic Laws of Flight.

Problem of Sustentation and Resistance of Wing Surface

A monoplane weighing 2000 pounds uses an R. A. F. 6 wing section.

(a) What area will it require so that its lowest speed may be 45 miles an hour?

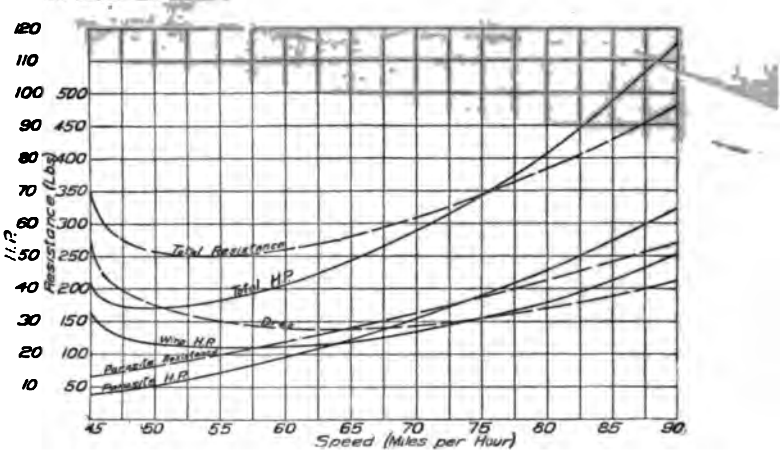


FIG. 10. RESISTANCE, HORSE-POWER AND SPEED DIAGRAM.

(b) What will be the drag of the wing at this speed and what will be the horse-power required for the wing alone?

(c) Assuming that the parasite resistance (resistance of the body, chassis, wires, struts, etc.) is 120 pounds at 60 miles an hour, and that it varies directly as the square of the speed, what will be the total resistance and horse-power required at this speed?

(d) If the power delivered at the propeller is 100 horsepower, what is the maximum speed available?

(a) Let A = wing area
W = weight of machine
D = drag of wing
P = parasite resistance
R = total resistance $= D + P$.

From Fig. 9 we see that the maximum value of K_y is .00309, at 16°. Then, since $W = K_y A V^2$, and $V = 45$ miles per hour,

$$A = \frac{W}{K_y V^2} = \frac{2000}{.00309 \times (45)^2} = 319 \text{ square feet.}$$

(b) From Fig. 9, L/D at 16° = 6.8. Then $D = \frac{2000}{6.8} = 294$ pounds.

Since $\frac{1}{375}$ horse-power is required to overcome a resistance of 1 pound at 1 mile per hour, horse-power $= \frac{DV}{375} = \frac{294 \times 45}{375}$ $= 35.3$ horse-power to overcome wing drag at 45 miles per hour.

The drag of the wings can also be obtained, of course, by substituting the proper value of K_x in the equation

$$D = K_x A V^2$$

The first method described will prove the simpler when a number of cases are to be worked out, but the second is more accurate at very small angles of incidence.

(c) At 60 miles an hour $K_y = \frac{W}{AV^2} = \frac{2000}{319 \times (60)^2} = .00174.$

Fig. 9 shows us that this value of K_y will be attained at an angle of incidence of 5.7°, at which angle $L/D = 14.2$.

Then $D = \frac{2000}{14.2} = 141$ pounds, and $R = D + P = 141 + 120 = 261$ pounds.

The power required equals $\frac{RV}{375} = \frac{261 \times 60}{375} = 41.8$ horsepower.

(d) In order to determine accurately the speed obtainable with a given power, it is necessary to plot a curve of power required at various speeds. In computing points on this curve, we assume the parasite resistance proportional to V^2. This is approximately true, the deviation being due to changes in resistance coefficients of body, struts, etc., as the angle at which they meet the wind changes. Proceeding on this assumption, $P = KV^2$, since $P = 120$ pounds when $V = 60$ miles per hour, $K = \frac{P}{V^2} = .0333$, and $P = .0333V^2$.

In Table 5 are given a few points on such a curve, computed as was that for 60 miles an hour, which we just secured. A student might carry through some of these computations, checking his results against those here given, in order to make sure that the method is perfectly clear to him.

TABLE 5.

V	K_y	Angle of inc.	K_x	L/D	D	P	R	Wing H.P.	Parasite H.P.	Total H.P.
45	.00309	16.0	.000446	6.9	289	67	356	34.8	8.0	42.8
50	.00251	9.9	.000218	11.5	174	83	257	23.2	11.1	34.3
55	.00207	7.4	.000156	13.3	150	101	251	22.1	14.8	36.9
60	.00174	5.6	.000122	14.3	140	120	260	22.4	19.2	41.6
65	.00148	4.4	.000102	14.5	138	141	279	24.0	24.4	48.4
70	.00128	3.5	.000093	13.8	145	163	308	27.1	30.4	57.5
75	.00111	2.8	.000087	12.8	156	187	343	31.2	37.4	68.6
80	.00098	2.2	.000082	11.9	168	213	381	35.7	45.5	81.2
85	.00087	1.8	.000081	10.7	187	241	428	42.5	54.6	97.1
90	.00077	1.3	.000081	9.5	211	270	481	50.6	64.8	115.4

References to Chapter 4

CONTROVERSIAL ASPECTS OF VALUES OF COEFFICIENTS FOR FLAT PLATES OF VARYING SIZES:

Notes on the Dimensional Theory of Wind Tunnel Experiments. E. Buckingham; Reports on Wind Tunnel Experiments in Aerodynamics, Smithsonian Miscellaneous Collections.

"Critical Speeds for Flat Disks in a Normal Wind," J. C. Hunsaker and E. B. Wilson; loc. cit.

Bulletin de l'Institut de Koutchino, Moscow, 1912.

DATA FOR FLAT PLATES.

"The Resistance of Air and Aviation," G. Eiffel, translated by J. C. Hunsaker.

DATA FOR R. A. F. 6 WING SECTION.

Reports on Tests of Four Aerofoils. Report of the British Advisory Committee on Aeronautics, 1912-1913. Report No. 72.

Chapter V

Comparison of Standard Wing Sections

The National Physical Laboratory has often been criticized in the past for not stating, in spite of its voluminous reports, what the "best" wing section is. There is no such thing as a "best" section. There are very bad wing sections giving abnormally high resistance and low lifting power; there are sections giving high lift at big angles of incidence, but too great a resistance at small angles, others giving a low maximum lift, but very suitable for high speeds; others give a very stable motion of the center of pressure, but sacrifice aerodynamic efficiency. The selection of any particular form depends on the performance required of the machine in view.

As the result of several years' practice, modern machines all tend to a few types of wings, although there are numberless small modifications by individual designers. We shall attempt to classify and give data for what may be called Standard Sections, using the (pounds, per square foot, per miles hour) system of units for force coefficients.

Representative Wing Sections Selected

These have all been taken from the N. P. L. and Eiffel laboratories. German laboratories have done a great deal of work with reference to propeller sections, and also have carried out tests on wing shapes of a great many forms, but the present selection is representative and sufficient for all practical purposes. When a designer wishes to introduce slight variations in the standard forms, it will be always necessary for him to submit his variation to a special test, so that a complete collection of every form that has ever been submitted to publication would be useless.

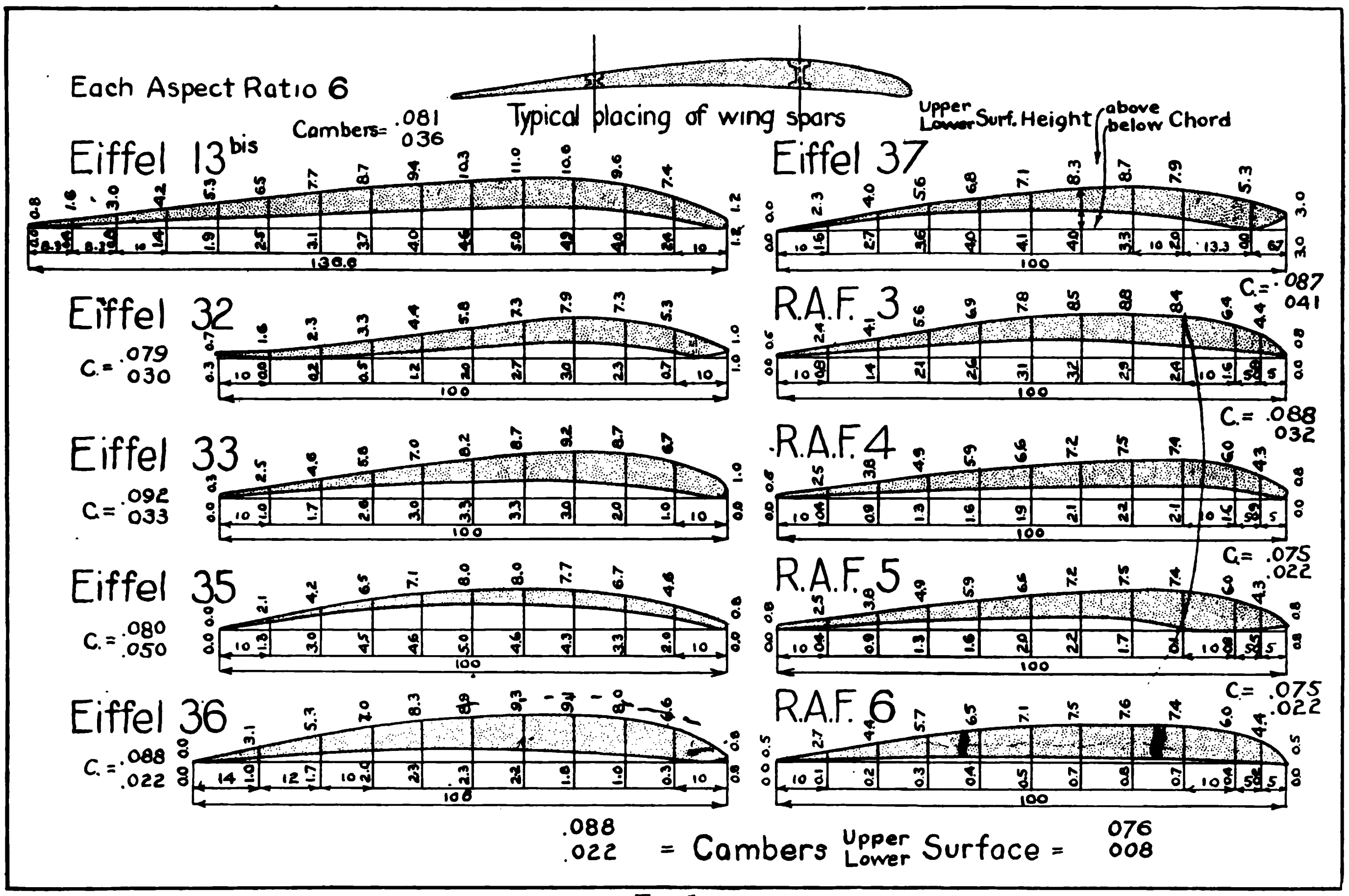


Fig. 1.

The sections we have selected are: R. A. F., No. 3, 4, 5, 6 and Eiffel 13 bis., 32, 33, 35, 36, 37. In Fig. 1 these forms are represented on a uniform plan, with complete dimensions, and values of camber. The camber of the upper surface is defined as ratio of maximum height above chord to chord length, and the same definition holds for the lower surface. The hollowing out of the lower surface, as we shall see later, has little importance —it scarcely affects the Lift/Drag ratio or the angle of incidence for the burble point, but it increases the lift about 17 per cent at any angle when a plane lower surface is cambered out to a camber of 0.06. An increase in lift obtained in this way involves a dangerous weakening of the wing. In Dr. J. C. Hunsaker's opinion, a decrease of camber below 0.05 or an increase of camber above 0.08 for the upper surface is disadvantageous in practice. Broadly speaking for the incidence

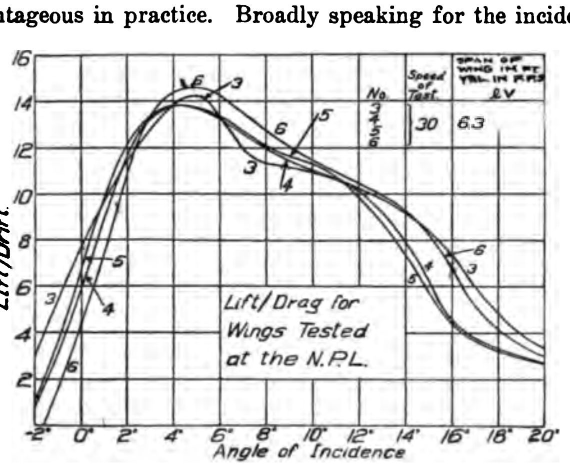


FIG. 2.

giving maximum Lift/Drag ratio, the lift for upper surface camber of 0.08 may be twice as great as for a camber of 0.05, but the Lift/Drag ratio is diminished by nearly 25 per cent. We shall deal later with the effects of varying the position of the maximum ordinate of the upper surface; the best position for this maximum ordinate is about 3/8 of the cord from the leading edge.

Complete Data Presented

In Figs. 2 to 8 are given curves for Lift, Drag, Lift/Drag

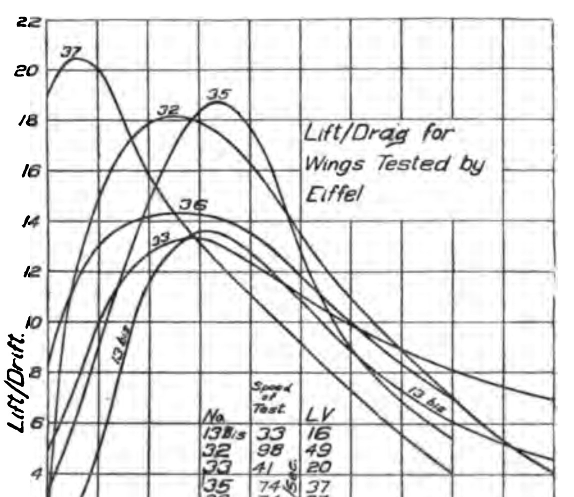


FIG. 3.

and Center of Pressure motion for these wings. In Fig. 9 a comparative table has been drawn up giving maximum Lift coefficients and corresponding angles; maximum L/D and corresponding angles; the angle of incidence and the corresponding L/D for a lift coefficient of value .00086, and also the value of V for the tests from which these results have been taken. This is as complete data as the designer can possibly require. The aspect ratio for all these sections is 6.

We shall deal later with the effects of variation of scale and speed. At this point it is sufficient to state that whereas the lift coefficient is unaffected by variation in the product lV—span of wing in feet times velocity of relative wind in feet per second—the drag coefficient and the L/D ratio are both improved by increase in lV. The N. P. L. tests and Eiffel's tests are unfortunately not concordant in this respect. Eiffel's experiments were made in a larger wind tunnel and at higher speeds, and if the same wing were tested at the N. P. L. and Eiffel's laboratory, the latter would give better results for both drag and L/D. Since in an actual machine the product lV will be very much greater than the values of either laboratory, the full size performance will always be somewhat better than

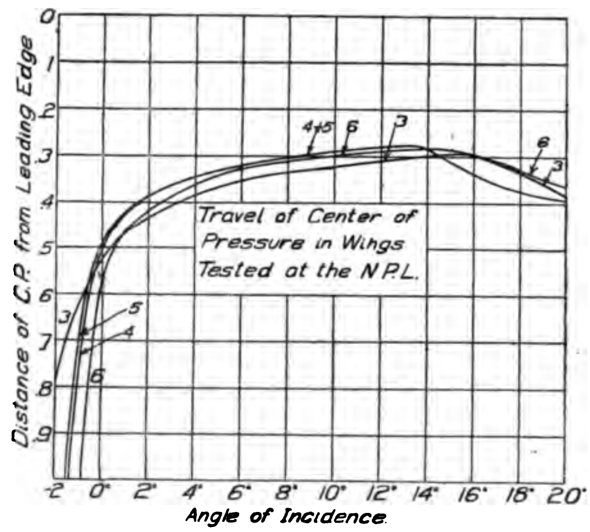


FIG 4.

the one deduced from these experimental results, particularly where an N. P. L. section is used. Employing the exact figures of our curves, the designer will be proceeding on a very conservative basis. Certain experiments of the N. P. L.—which we shall deal with fully later—permit us to make approximate corrections. These have been made in the last column of Fig. 9.

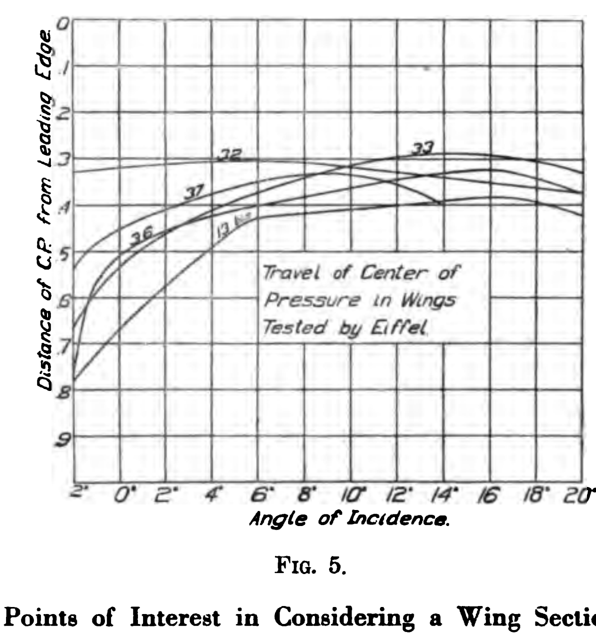


FIG. 5.

Points of Interest in Considering a Wing Section

In discussing the merits of a section, there are so many points at issue that it is only in an actual design that it is possible to enter fully into all. Study of the data submitted will be of much more use if the following features are always kept in mind:

(a) The maximum value of L/D, and the corresponding K_y.

A machine in normal horizontal flight will generally be navigated at the angle giving the best L/D ratio, which is therefore most important from an efficiency point of view. The

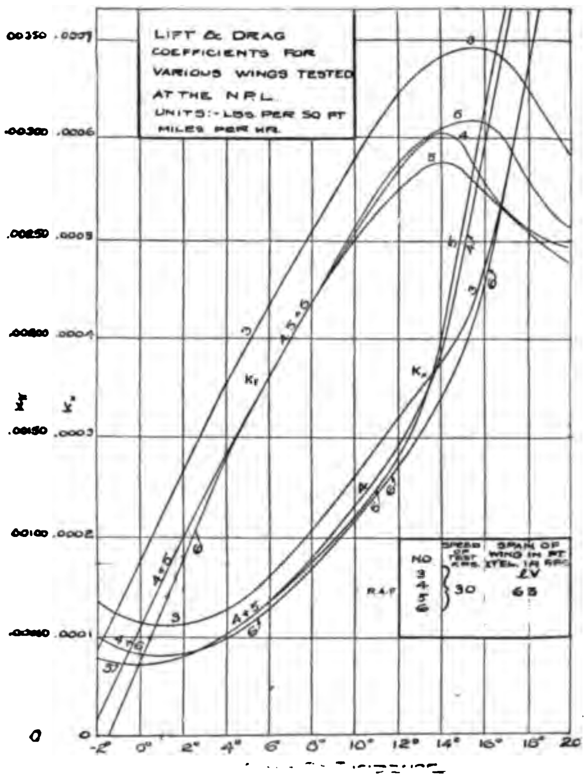


FIG. 6.

value of the lift coefficient at the best L/D is of importance. The greater the lift at this ratio the smaller the area of the wing surface required for a given load. With a heavy machine, such as a flying boat, or an armored battleplane, a big

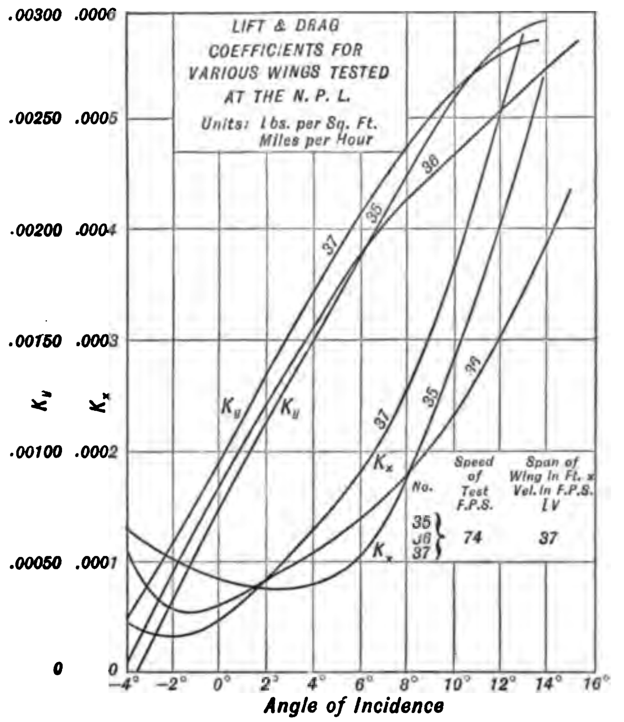


FIG. 7.

lift coefficient is essential. With a speed scout or a light reconnaissance machine, a small value of K_y at best L/D is usual. With a sufficiently powerful motor a small wing surface may be used and a great speed attained.

(b) The maximum K_y has a bearing on a number of points. The greater the maximum K_y the slower is the speed at which a machine may land. If the maximum K_y, or simply large values of K_y, are accompanied by a good L/D ratio, then the machine will be efficient and ready in climbing—though the best angle of climb is by no means the angle of maximum K_y, as we shall see later in considering the economic laws of flight.

(c) The maximum K_y should occur at as high an angle as possible, so as to give a big range, and possibility of a large speed variation.

(d) The angle of maximum lift is termed the burble point,

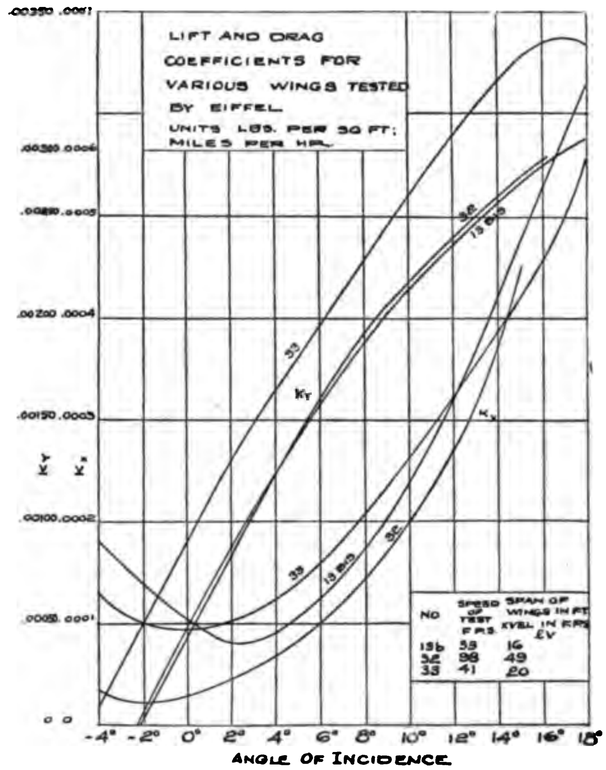


FIG. 8.

as we know, and also the "stalling" angle. It is very important to consider what the shape of the lift curve is in the neighborhood of this angle. If the lift past the burble point falls off very rapidly, the pilot may easily stall the machine. He may increase the angle of incidence too far and find his sustaining power fall off dangerously. A wing with a flat lift curve at the burble point will avoid such danger.

(e) The L/D ratio at small angles of incidence and small values of K_y determines whether the machine is really suitable for high speeds. We have arbitrarily chosen $K_y = 0.00086$ as the value of comparison, and it can be seen from the tables how widely L/D varies at this point. A machine with good maximum L/D and a high maximum K_y might be totally inefficient at high speeds.

(f) The movement of the center of pressure is important at low angles. If at low angles the center of pressure moves steeply back towards the trailing edge, the machine will have a tendency to "dive," provided for, of course, by fixed stabilizing surfaces on modern machines. If the center of pressure remains stationary, on the other hand, as in Eiffel 32, it will maintain its attitude at low angles, and will not tend to dive even with small stabilizing surfaces and inefficient or inoperative elevator. Similar considerations apply to "stalling" angles.

(g) In addition to the separate consideration of these points, there yet remains the appraisal of the wing through-

FIG. 9 COMPARATIVE TABLE OF STANDARD WING SECTIONS

LIFT COEFFICIENTS IN LBS. PER SQ. FT.; MILES PER HR.

ASPECT RATIO 6

Span of Wing in Feet x Relative Wind in Ft/Sec. lV.	Camber		Wing	Max. K_y			Max. L/D			K_y = .00086		Max. L/D —corrected app. to full size mach. in actual flight in accordance with exp'ts at the N. P. L.
	Upper	Lower	Section	Angle	K_y	L/D	Angle	K_y	L/D	Angle	L/D	
16	0.081	0.036	Eiffel 13 Bis.	.....	.....	....	4.3°	0.00129	13.6	1.9°	11.4	14.5
49	0.079	0.030	" 32	.....	.....	....	3.0°	0.00103	18.2	2.2°	18.0	18.2
20	0.092	0.033	" 33	16.8°	0.00336	7.2	3.5°	0.00152	13.4	−0.2°	9.5	13.6
37	0.080	0.050	" 35	14.6°	0.00296	5.2	4.8°	0.00165	18.7	0.5°	10.6	18.9
37	0.088	0.022	" 36	.....	.....	....	3.1°	0.00142	14.3	0.0°	13.1	14.3
37	0.087	0.041	" 37	14.1°	0.00288	4.0	−0.8°	0.00086	20.4	−0.8°	20.4	20.4
6.3	0.088	0.032	R.A.F. 3	15.7°	0.00347	7.8	5.0°	0.00195	14.3	−0.1°	7.4	18.1
6.3	0.075	0.022	" 4	14.0°	0.00304	8.0	4.2°	0.00142	13.8	1.4°	10.3	19.2
6.3	0.075	0.022	" 5	14.2°	0.00288	7.0	4.2°	0.00142	13.8	1.4°	11.0	19.2
6.3	0.076	0.008	" 6	15.4°	0.00310	7.8	4.9°	0.00157	14.6	1.9°	10.4	18.5

out its performance. The designer must see how far one point of excellency conflicts with other requirements; what the range is. The ideal wing would give great lift and efficient climb, high efficiency in normal flight, and high efficiency at maximum speeds.

(h) A wing may be entirely satisfactory from an aerodynamic point of view, and yet fail to satisfy as regards structural requirements. In Fig. 1 is shown a typical arrangement of the wing spars. It is important that the points where the wing spars are likely to be placed, the wing should have sufficient thickness to permit the use of reasonably deep spars without exaggerated width. A wing may indeed have sufficient thickness at two points for good spars to be placed, yet these points may be totally unsuitable. They may be too near together, so that a weak overhanging construction or excessive spar loading is the result, or too far apart so that too long an unsupported rib section results.

There could be no better plan for the reader to whom the subject is comparatively new than to go through all the wing sections presented with reference to these eight points.

Consideration of a Few Sections in Common Use

We shall consider a few sections in this manner ourselves.

Take Eiffel 37, for instance. Its maximum L/D—the highest of any section considered here is 20.4, occurring at —0.8°, which is still a good many degrees from the angle of no lift, though its center of pressure motion at this point is rapid. Its maximum K_y is small (0.00288), with a L/D of only 4.0. Such a machine would be unsuitable for heavy loading, but would be excellent for a high speed racing machine, in which little variation in speed would be required. It would, however, have to land at comparatively high speed because of the low maximum K_y. The structural difficulties would be considerable, because there is insufficient thickness in the wing for the rear spar.

Eiffel 32 is an excellent all-around wing. Its maximum L/D, uncorrected for scale, is high 18.7. It has fairly good values of L/D for high lift coefficients. Its center of pressure motion is almost nil.

R. A. F. 3 has the highest value of K_y (0.00195) at maximum L/D. It would be suitable for a heavy flying boat. At small values of K_y, on the other hand, its L/D is very small. It would be unsuitable at high speeds. Structurally it is excellent.

R. A. F. 6 would also be a good all-around wing, not capable of sustaining the heavy loads of R. A. F. 3, or given the high speed of Eiffel 37, but compromising usefully.

References for Chapter 5

Eiffel: "Nouvelles Recherches Sur la Resistance de l'Air et l'Aviation."

British Report, 1912-1913. No. 72. Report on the Results of Tests of Four Aerofoils.

Chapter VI

Effects of Variation in Profile and Plan Form of Wing Sections

As we have seen in Chapter 5, numberless variations are possible in the profile of wing sections. A slight variation in the profile may, however, introduce considerable changes in the aerodynamic properties of a wing, and necessitate a wind tunnel test. Experiments conducted at the various laboratories on variations of camber, of position of maximum ordinate, on the thickening of leading and trailing edges, and so forth, have therefore rather a qualitative than a quantitative significance. But the results obtained deserve attention, and may serve as a guide to useful modifications. The most important of these experiments are summarized here, and a fuller reference list is appended.

Effect of Variation of Position of Maximum Ordinate in a Wing Section of Plane Lower Surface, and Constant Camber 0.100 for Upper Surface

These experiments of the N. P. L. are mainly interesting because they indicate where approximately the maximum ordinate of a section should be to give the best possible L/D ratio.

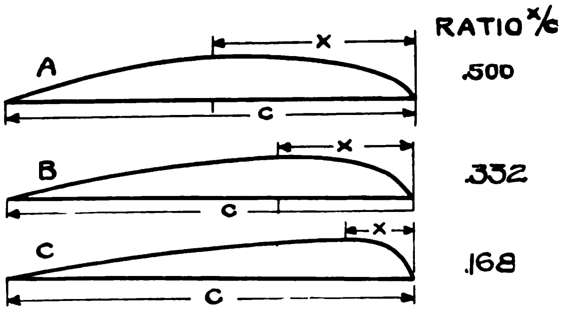


FIG. 1. SECTIONS USED IN INVESTIGATING VARIATIONS OF POSITION OF MAXIMUM ORDINATE

In Fig. 1 are shown a selection of three of the sections tested. They were all developed from one section by altering the position of the maximum ordinate and compressing or expanding the other ordinates to correspond. The Lift and Lift/Drag curves for these sections show considerable variations in values as can be seen from the following table:

TABLE 1.

WING SECTIONS PLANE LOWER SURFACE. UPPER SURFACE CAMBER 0.100. POSITION OF MAXIMUM ORDINATE VARIED.

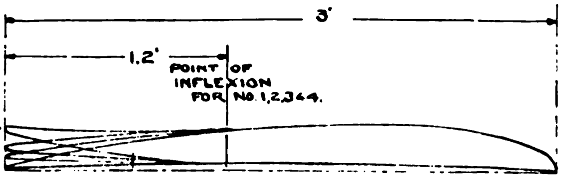
Section.	Ratio of position of maximum ordinate to chord length.	Maximum L/D.	Angle for maximum L/D.	Maximum K_y in lbs., sq. ft., miles/hour. units.	Angle for maximum K_y
A........	.500	11.2	8°	.00317	18°
B........	.332	13.6	4°	.00358	16°
C........	.168	11.0	4°	.00206	8.5°

We see that the maximum L/D for section B with a ratio .332 is as high as 13.6, while for section C, where the maximum ordinate is well forward, it sinks to 11. Again, the maximum lift for B is about 50 per cent greater than that for C. The angle of maximum lift also appears much earlier when the maximum ordinate is nearer the leading edge. A further inspection of the N. P. L. curves also shows that at the point of maximum lift, a slight variation in the ratio changes a smooth burble point into a dangerously steep one.

The main result of the investigation is to show the care required in altering even slightly the position of maximum ordinate for a given section, and also to indicate that the best position is about one-third from the leading edge.

Behavior of Wings with Reverse Curvature at the Trailing Edge

This constitutes a far more important question than that of the preceding paragraph. It would considerably simplify



NO. OF TAIL	1	2	3	4
Amount tail is raised as a fraction of chord	.000	0.011	0.027	0.057

FIG. 2. MODIFICATIONS OF THE R. A. F. 6 WITH UPTURNED TRAILING EDGES

airplane design, from the point of view of statical and dynamical stability if the position of the centre of pressure or of the vector of resultant force on the wing did not vary its position so rapidly with change in the angle of incidence. It may be said that as a general rule for the usual angles of flight that when the angle of incidence decreases the centre of pressure on a wing moves far back, and the resultant force tends to dive the machine, decreasing the angle of incidence still further. When the angle of incidence increases, the centre of pressure moves forward and the resultant force tends to stall the machine, increasing the angle of incidence still further. We shall deal fully with this important point when considering the general statical equilibrium of the airplane.

Among other means of attaining stability, wings have been designed with a slight reverse curvature at the trailing edge, which have been very successful in keeping the centre of pressure motion within narrow limits. It is important to us to see what sacrifice of sustaining power and efficiency reverse curvature entails.

At the N. P. L. a section (No. 1) very similar to that of the R. A. F. 6 was employed, and three reversed curvature forms 2, 3, 4 were developed from it by turning up the trailing edges through successively increasing distances while keeping the thickness of section unaltered. The point of inflexion, at which the reflexing began was in each case 0.4 of the chord from the trailing edge, though this could be varied to 0.2 without much effect. These sections are illustrated in Fig. 2.

The travel of the centre of pressure is shown in Fig. 3 for

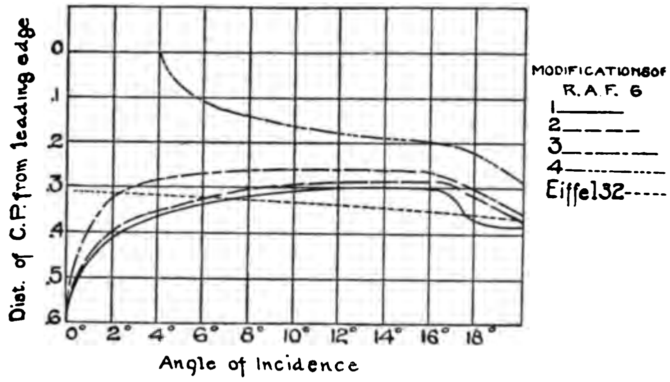


FIG. 3. TRAVEL OF CENTER OF PRESSURE FOR A SERIES OF WINGS WITH UPTURNED TRAILING EDGES

all five sections. The curves for the N. P. L. sections show that as the elevation of the trailing edge increases, the centre of pressure motion becomes less marked in its movement toward the trailing edge, than stationary, and finally moves toward the leading edge. This is certainly satisfactory from the stability point of view, but the questions of efficiency and maximum lift have also to be considered. The following are the values obtained for maximum L/D and maximum K_y:

TABLE 2.

Section.	Amount tail is raised as fraction of chord.	Maximum L/D.	Maximum K_y
1	0.000	15.7	.0320
2	0.011	15.0	.0294
3	0.027	14.3	.0282
4	0.057	13.0	.0245

It can be seen that as the rear edge is turned up the L/D and the maximum K_y both decrease progressively.

The main conclusion of the British investigators was that with an elevation of the rear edge of about .037 of the chord, the centre of pressure can be kept stationary, but with a loss of 12 per cent. of the maximum L/D and 25 per cent. loss of the maximum possible lift. This would be too great a sacrifice for the sake of stability and the designer would find other methods of stabilization such as the use of décalage in biplanes and negative stabilizers far more useful.

Eiffel has, however, investigated a section with a very slightly reversed trailing edge (Eiffel No. 32 Lanier-Lawrance, details of which have been given in Chapter 5), which is far more satisfactory and in wide use. Its maximum L/D is about 18.2, maximum lift coefficient is about .0033, and it has an excellent working range. The centre of pressure motion is almost nil between 0 degrees and 10 degrees of incidence, and such a wing would certainly not tend to dive a machine, although it is not very good at stalling angles. Its shape offers certain constructional difficulties in the region of the rear spar.

Effect of Thickening the Leading Edge of a Wing

Contrary to a somewhat common conception, the thickening

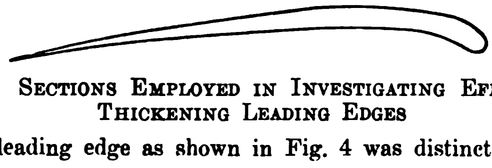


FIG. 4. SECTIONS EMPLOYED IN INVESTIGATING EFFECTS OF THICKENING LEADING EDGES

of the leading edge as shown in Fig. 4 was distinctly disadvantageous, the decrease in efficiency progressing proportionately to the thickening.

Effects of Thickening Wing Towards the Trailing Edge

Thickening towards the trailing edge is sometimes advantageous from the point of view of structural strength, and experiments have been conducted to see the loss in aerodynamic efficiency such thickening involved. The sections em-

FIG. 5. EXPERIMENT OF THICKENING THE TRAILING EDGE OF WING

ployed are shown in Fig. 5. It appears from these experiments that the lift coefficient at a given angle of incidence is not much affected at angles greater than 7 degrees but that at smaller angles of incidence the lift coefficient is actually a little greater for the thickened sections. The maximum Lift/Drag steadily diminishes as the trailing edge is thickened:

TABLE 3.

Section.	Maximum L/D.
1	13.2
2	13.4
3	14.2
4	14.6

"Phillips Entry"

As shown in Fig. 6, the section R. A. F. 4 was modified into the R. A. F. 5 to give the well-known "Phillips Entry." This

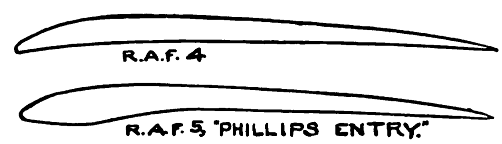


FIG. 6. MODIFICATION OF R. A. F. 4 WING TO GIVE PHILLIPS ENTRY

modification was found to have no effect on the aerodynamic properties of the wing, an important consideration in view of the fact that numerous attempts have been made to utilize this modification.

Effects of Varying Aspect Ratio

Föppl's and Eiffel's experiments have dealt with cambered plates; the N. P. L. has investigated the effect of varying aspect ratios on a practical wing section rectangular in plan

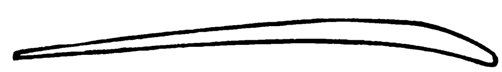


FIG. 7. WING SECTION EMPLOYED AT THE N. P. L. IN INVESTIGATION OF EFFECTS OF VARYING ASPECT RATIO

similar to the Bleriot XI bis which is shown in Fig. 7. For a more or less accurate understanding of the phenomena accompanying such variation, it is necessary to consider pressure distribution, but for design it is more important to bear in mind the simple results of this investigation:

As aspect ratio increases

(1) The maximum L/D ratio improves, the corresponding angle of incidence remaining sensibly the same, and the L/D at other angles improves also.

(2) the drag diminishes.

(3) the lift coefficients at all except very small angles and the maximum lift coefficient remain practically constant; the

maximum lift coefficient occurs at a smaller angle of incidence.

(4) the angle of no lift occurs at smaller positive angles, or larger negative angles as the case may be.

Although the Bleriot wing tested by the N. P. L. was of practical form, it is not commonly employed in modern construction. The correction tables (Tables 4 and 5) are solely based on results derived from it, and it does not at all follow that similar corrections would apply to wings of other form. In default of other experimental work, however, such corrections can be applied with probably a fair degree of accuracy. The values for aspect ratio of 6 are taken as a standard of comparison, this being the aspect ratio used for so much experimental work on wing sections.

TABLE 4.

APPROXIMATE CORRECTIONS FOR MAXIMUM L/D WITH VARIATION OF ASPECT RATIO.

Aspect ratio.	Ratio of maximum L/D to maximum L/D at aspect ratio 6.
3	.72
4	.82
5	.92
6	1.00
7	1.08
8	1.11

The following table shows the ratio of drag for various aspect ratios to drag for aspect ratio 6 as unity:

TABLE 5.

APPROXIMATE CORRECTION FOR VALUES OF K_x WITH VARIATION OF ASPECT RATIO.

Angle of incidence.	Aspect Ratio. 3	4	5	6	7	8
0	1.12	1.05	1.00	1	1.10	1.00
2	1.15	1.90	1.02	1	1.00	1.05
4	1.13	1.022	1.10	1	1.00	1.00
6	1.11	1.031	1.03	1	.91	1.14
8	1.22	1.040	1.01	1	.89	.91
10	1.04	1.047	1.06	1	.88	.91
12	1.30	1.056	1.11	1	.98	.99
14	1.14	1.071	1.02	1	.92	.89
16	1.17	.94	1.00	1	.85	1.01
18	.876	1.130	.91	1	1.05	1.20

Choice of Aspect Ratio

In selecting ratio for an airplane many other considerations enter besides those of aerodynamic efficiency. Thus as aspect ratio and the span of the wings increase, the heavier the structure becomes for the same strength. This involves heavier bracing and more structural head resistance; the increase in weight itself reduces the aerodynamic efficiency indirectly. Hence if the aspect ratio were increased to an exaggerated extent, structural difficulties would more than counterbalance the gain due to this increase. The question is too complex for theoretical treatment or for definite rules. Later in the design of a standard machine, comparative designs will be made for various values of aspect ratio.

For preliminary design, the best method of fixing aspect ratio is to follow standard practice, and this would indicate:

5 to 1 aspect ratio for monoplanes and small biplanes.

6 to 1 or 7 to 1 for large biplanes.

Effects of Raking the Plan Form of a Wing

Experiments on the effect of raking the plan form of a wing have been conducted by Eiffel in France and Föppl in Germany, references to which are given at the end of this section. Unfortunately, their investigations were mostly on circular wings, were somewhat contradictory, and their results varied with different cambers.

In the experiment which Eiffel conducted on a practical wing section, Coanda Wing, Eiffel No. 38, as illustrated in Fig. 8, the raked wing was decidedly superior to the rectangular form into which it was cut down. Nor can this improvement be due to variation in aspect ratio which is negligibly small. The ratio of maximum L/D was about 1.2 to 1.

It would seem therefore that experiment is in agreement with practice in imputing certain advantages to raking. But in view of the variation in results with wings of different

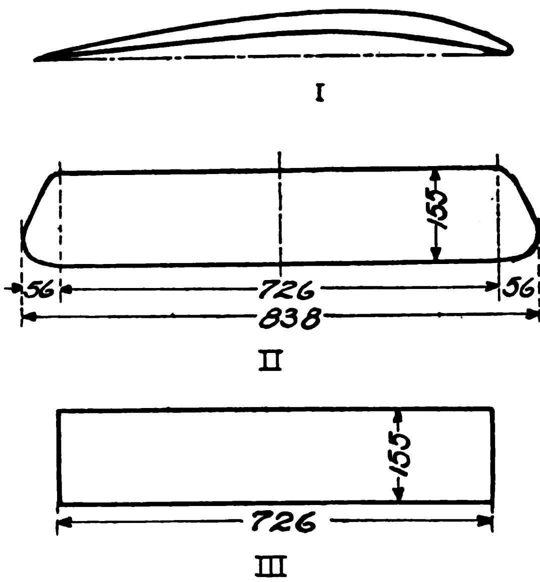


FIG. 8. SECTIONS EMPLOYED IN EIFFEL'S EXPERIMENTS ON "RAKING"

camber, it would be unsafe to employ a correction ratio of 1.2 in maximum L/D for the raking of any other wing, say an R. A. F. 6 section, until there has been further investigation of this point.

Swept Back Wings

Another variation in the plan form of wing sections, very largely employed on German machines of recent type, and also on one or two American machines, is that of swept back wings. Swept back wings are mainly used to give lateral stability. It has also been thought that their arrow-like form gave them an increased aerodynamic efficiency, and that longitudinal stability was also improved by their employment. We are not at present concerned with lateral stability. Aerodynamically a recent investigation at the Massachusetts Institute of Technology shows a progressive decrease in efficiency with increased sweep back. As regards longitudinal stability the ac-

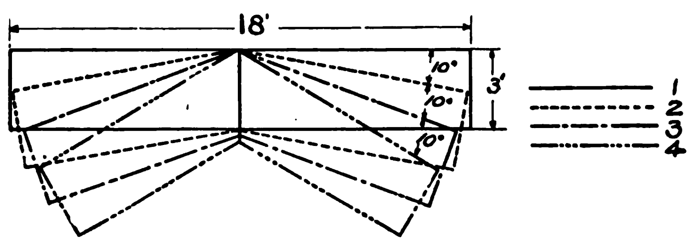


FIG. 9. WINGS USED IN EXPERIMENTS ON SWEPT BACK WINGS

tion is peculiar and not at all so satisfactory as that of the wings with reversed trailing edges.

An R. A. F. 6 wing, originally of aspect ratio 6 was employed and swept back as shown in Fig. 9. The results of the investigation are summarized in Table 6:

TABLE 6.

Section.	Sweep back.	Angle of incidence for maximum L/D.	Maximum L/D.	K_y for maximum L/D.	Angle of incidence for maximum K_y.	Maximum K
1	0	4°	17	.00143	14°	.00288
2	10	4°	16.5	.00130	16°	.00276
3	20	4°	16.2	.00129	16°	.00278
4	30	4°	12.8	.00120	[illegible]	[illegible]

Up to 20 degrees sweepback, it can be seen that the loss in efficiency is not so great, but the 30 degree entails a loss for which good lateral stability would scarcely compensate.

TRAILING EDGE

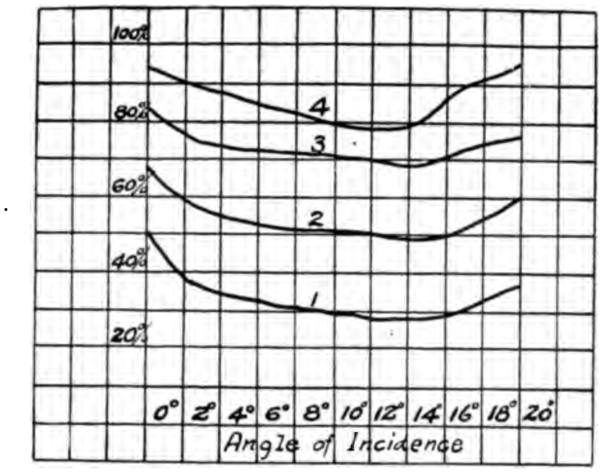


FIG. 10. MOVEMENT OF CENTER OF PRESSURE FOR WINGS WITH VARYING DEGREES OF SWEEP BACK

The centre of pressure motion is illustrated in Fig. 10. It has the same peculiar characteristic for each of the wings. At small angles the centre of pressure moves backward, thus producing diving, but at large angles the centre of pressure moves forward, thus tending to stall the machine. Longitudinal stability is thus not secured.

Negative Wings Tips of Swept Back Wings; Effect on Longitudinal Stability

Swept back wings with negative wing tips have been successfully employed in German machines; and in the Burgess-Dunne, without the use of tail surfaces. Such wings certainly give a great degree both of longitudinal and lateral stability, but at some sacrifice of efficiency. Experimental results, except for complete airplane models, are not available, but a simple theoretical discussion at this stage is instructive; this involves the application of the first principles of mechanics, yet always presented considerable difficulty. It also gives us the opportunity of considering the stabilizing influence of tail surfaces in an elementary manner.

Consider the two arrangements of Fig. 11, A and B, one

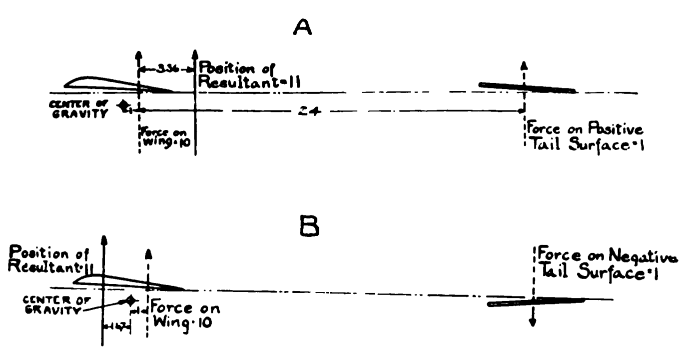


FIG. 11. DIAGRAM TO ILLUSTRATE VARIATION OF RESULTANT FORCE WITH POSITIVE AND NEGATIVE TAIL SURFACES

with a positive tail surface, the other with a negative tail surface. We will assume the forces on the wing and on the tail to be vertical for simplicity's sake, although this would not actually be the case, with positions of forces and centre of gravity as in sketch. Assume the force on the wing to be 10 times that on the tail. Then in case A moments about centre of gravity are:

$(10\times1)+(25\times1)=35$ in a diving or counter clockwise direction. The resultant must be aft of the centre of gravity, and since its value is $10+1$, it is $\frac{35}{11}=3.18$ feet aft of the centre of gravity between the two forces on wing and tail.

For case B moments about centre of gravity are:

$(10\times1)-(25\times1)=-15$ in a stalling or clockwise direction. The resultant will now be $\frac{15}{9}=1.67$ feet forward of the centre of gravity and forward of the force on the wing.

A negative tail can thus convert a diving moment into a stalling moment at small angles. At large angles of incidence the negative lifting surface will become positive and may be used to convert a stalling moment into a diving moment. A negative tail surface can thus be suitably adjusted to give longitudinal stability at all angles within the flight range.

Similarly for a machine with swept back wings and negative wing tips, as shown in Fig. 12, at an angle of 1 degree incidence for a positive section A-A, the force has a counter-

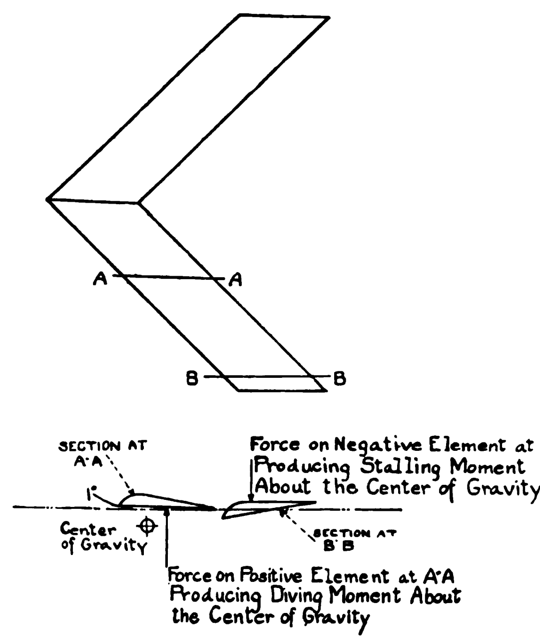


FIG. 12. DIAGRAM TO ILLUSTRATE STABILIZING EFFECT OF SWEPT BACK WINGS WITH NEGATIVE WING TIPS

clockwise moment about the centre of gravity tending to dive the machine. For a negative section B-B, the force has a stalling moment about the centre of gravity which prevents diving action. Similarly, at large angles of incidence the positive surfaces of the wing may tend to stall the machine, while the negative wing tips then assume a positive action and counteract the tendency to stall. Thus if the wings are sufficiently swept back and the negative surfaces powerful, static longitudinal stability can be secured.

The negative surfaces having so small an arm compared with negative tail surface must have a much larger surface than the latter. Consequently such an arrangement must be aerodynamically inefficient. This may be compensated for by the fact that no structural extensions to tail surfaces are necessary in a machine of this type.

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Chapter VII

Study of Pressure Distribution

For general purposes, a knowledge of lift, drag and the position of the vector of resultant force at various angles of incidence is as much aerodynamical data as the designer requires with reference to a wing section. But an investigation of pressure distribution bears directly on an understanding of the following important points:

1. The variation of stresses in the covering fabric of a wing due to the unequal distribution of pressures.
2. The great efficiency of a cambered surface as compared with a flat plate.
3. The analysis of the forces at play and their exact bearing on efficiency, and on the position of the resultant vector.
4. The relative importance and the inter-dependence of the two surfaces of a wing.
5. The effects of varying aspect ratio.
6. The variation of lift and drift with speed and size of model.

It is evident, therefore, that the question is not of purely scientific or academic importance. Much useful work has been done in this direction by Eiffel and the N. P. L., and a great deal still remains to be done.

Methods of Obtaining Pressure Distribution

The mapping of pressure distribution is a lengthy process requiring numberless readings. It is fully described in the N. P. L. reports, and we shall only summarize briefly the methods employed.

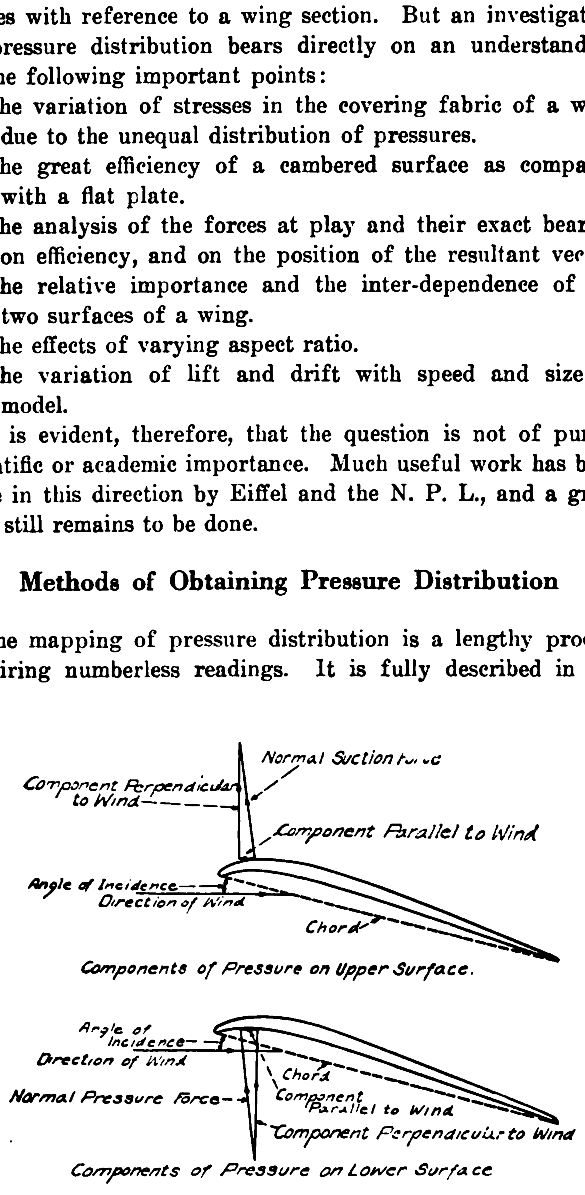


FIG. 1. DIAGRAM TO ILLUSTRATE HOW THE FORCES ON UPPER AND LOWER SURFACES ASSIST EACH OTHER.

Holes of $^1/_{64}$ inch diameter are drilled in the wing where required, normally to its surface, and are plugged with plasticine, except the one under observation. The hole in use is connected by a length of very thin hypodermic syringe tubing, too small to cause disturbance, with a three-way cock. A pitot and static pressure tube is placed in the channel where the flow is undisturbed by the presence of the model. The static pressure tube is permanently connected to one arm of the usual manometer; the other arm can be connected alternately by means of the three-way cock either to the pitot tube or to the hole drilled in the wing section.

The manometer can be thus made to read either the velocity head of the wind, or the difference in pressure between the static pressure of the channel, and the pressure on the wing at the point considered; and a direct comparison between these two quantities is immediately possible. Great care has to be exercised in obtaining values of pressure distribution which correspond to a constant value of the wind velocity, and in maintaining the same direction of the pitot tube relative to the wing.

Over the upper surface of a wing there will be suction, on the lower surface pressure, and we shall indicate the exact distribution in this section. In Fig. 1, the suction force normal to the upper surface, and the pressure force normal to the lower surface, are represented diagrammatically for the same position on the chord. If these forces are resolved along the line of the relative wind and perpendicular to it, we see that they add up to give a force upwind and lift. At other points along the chord these forces may oppose one another or give a force downwind. An elaborate method of graphical integration for pressure forces has been devised by the N. P. L., but their integration was normal and perpendicular to the chord. Such summations, if taken as giving lift and drag, involve errors except at very small angles.

Comparison of Results from Pressure Distribution and from Force Experiments

In Chapters 2 and 3 we have divided the forces acting on a wing into two classes: density or turbulence forces, and skin friction forces. A study of pressure diagrams en-

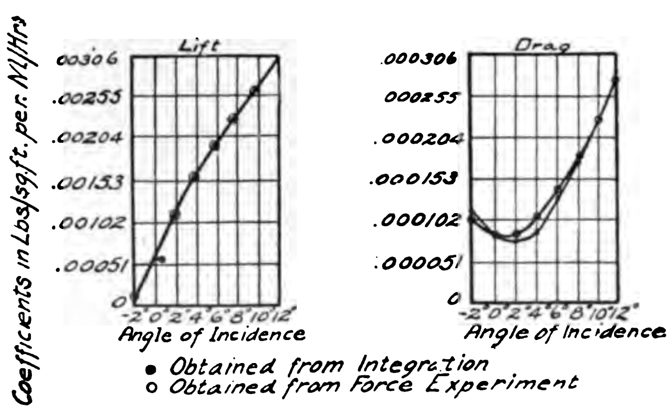


FIG. 2. COMPARISON OF RESULTS FROM PRESSURE DISTRIBUTION AND FORCE EXPERIMENTS FOR A R. A. F. 6 WING.

ables us to determine the part which these forces play in producing lift and drag. In Fig. 2, the results of a pressure force integration are shown from experiments on a R. A. F. 6 wing, and compared with the usual force determinations.

The result for the lift values coincides. Now, it is fairly clear that skin friction forces would not impart lift to a wing. We can conclude that lift is solely due to the density or turbulence forces, and, further, that lift can be obtained from the integration of the components of the pressure forces perpendicular to the relative wind.

The drag curves coincide at high angles, while between 0 degree and 8 degrees the drag derived from the pressure integration is less than that obtained from force experiments. The difference is due to the fact that the pressure results give no indication of skin friction forces.

Effect of Variation of Speed and Scale on Lift and Drag Coefficients

These considerations enable us to deal more closely with the question of variation in coefficients with change of speed and scale, the product (lV).

Experiments at the N. P. L. show that lift coefficients are scarcely affected by such change. If, as has been shown, lift is due to pressure forces solely, there is no reason why the lift should be affected.

That portion of the drag due to density resistance and accounted for by the pressure integration would vary as AV^2. But the skin friction, not accounted for by the pressure experiments, varies as $bl^{.03}V^{1.86}$. Therefore, with increase in speed and scale, the drag would not vary as AV^2, but somewhat less rapidly, and K_x would not be a constant.

It might be possible for any model wing to find the density component of the drag by allowing for skin friction, step this

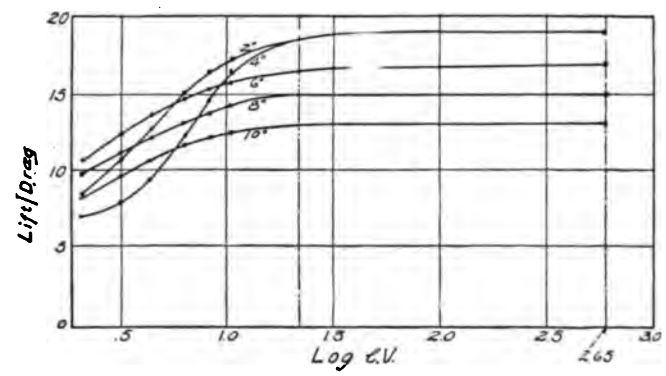


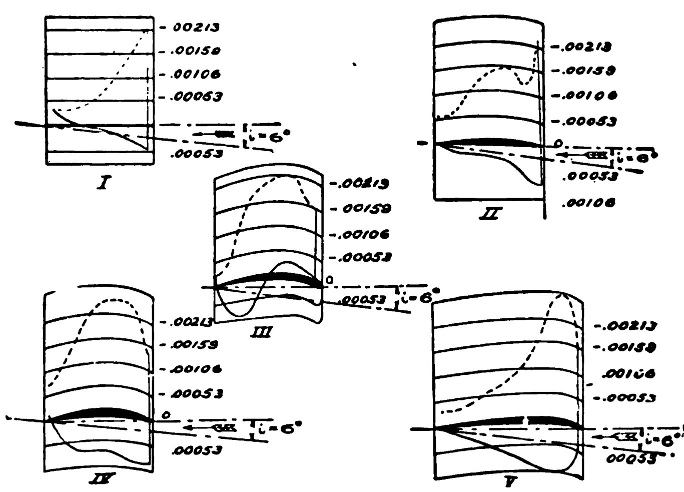
FIG. 3. DIAGRAM SHOWING VARIATIONS OF LIFT/DRAG WITH lV FOR R. A. F. 6.

up to full size as AV^2 and then to compute the skin friction from the $bl^{.03}V^{1.86}$ formula. But we are not too sure of this formula, and the process would be very complicated.

Some experiments at the N. P. L. provide, perhaps, the best guide, although they have been carried over too narrow a range. In Fig. 3 are given values L/D for the R. A. F. 6 section, plotted at the same angle of incidence against log (lV), the logarithm being used purely for convenience in plotting. If the designer wishes to correct to the full-sized machine, he must take the value lV which is given in connection with the section he employs and compare values of L/D at any angle of incidence with that corresponding to log $lV = 2.75 (lV = 560)$, a good value for a full-sized machine. The N. P. L. also gives corrections for lift, which are to be regarded with doubt, and corrections for drag coefficients. But it would seem safer to employ only the L/D corrections; and even these should be only used for the designer's personal benefit, or in comparing the merits of two sections, as in the last column of Fig. 9 of Chapter 5. Designing without any corrections would be the most conservative method, and might ensure a pleasant surprise for full-size performance.

Distribution of Pressure at Median Cross Section of Various Surfaces

With the ribs, stringers, fillers and good fabrics employed in modern wing construction, the stresses produced in the fabric are well within the limits of safety, as we shall see later. But it is important to remember that it is not the mean pressure over a wing which gives the maximum stress in the fabric; it is the maximum pressure at one particular point. Also a small hole at one point of the fabric may



	No	Maximum effect of suction above plus pressure below	Position of maximum effect as fraction of cord from leading edge	Pressure per square foot at 60 mph.
Flat Plate	I	.00266	1/15	8.59
Eiffel 7	II	.00309	1/2	11.3
Eiffel 9	III	.00245	2/3	8.82
Eiffel 8	IV	.00340	2/5	12.2
Eiffel 13	V	.00372	1/7	13.4

All coefficients in pounds per square foot per foot second.

FIG. 4. DISTRIBUTION OF PRESSURE FOR MEDIAN CROSS-SECTIONS FOR VARIOUS SURFACES AT 6 DEGREES ANGLE i.

cause it to carry the added effect of the suction at the upper surface and the positive pressure at the lower surface.

Eiffel in his earlier experiments consistently investigated the pressure distribution over both surfaces, and a number of his diagrams are shown in Fig. 4, while the maximum effect of suction on upper surface and pressure on lower face is shown in table in Fig. 4, at the same angle of incidence of 6 degrees in each case. The speed of the test was 32 feet per second. Although these surfaces are not in common use, they serve as a qualitative criterion from this standpoint for more practical wings.

An incidence of 6 degrees may be taken as normal flight, and at a speed of 60 miles per hour it is seen that pressures may vary from 13.4 to 8.82 pounds per square foot. This is, however, by no means the worst loading that can occur on a wing fabric. Under abnormal conditions such as flattening out after a steep dive, the maximum load per square foot may be many times greater. This question will be considered in detail in dealing with factors of safety.

Distribution of Pressure Over the Entire Surface of a Wing; Lateral Flow, Its Bearing on Aspect Ratio

The most instructive experiments on the pressure distribution over the entire surface of an aeroplane are those due to Jones and Patterson, at the N. P. L. and to Eiffel.

At the N. P. L. a single wing section has been dealt with in this way, but there has been a close and useful analysis, which is a first step in the investigation of phenomena of lateral flow, and of the underlying causes of the effects of varying aspect ratio.

A section resembling the R. A. F. 6, but with somewhat greater camber, was employed. Rectangular in plan, it had a series of observation points as shown in Fig. 5, on five sec-

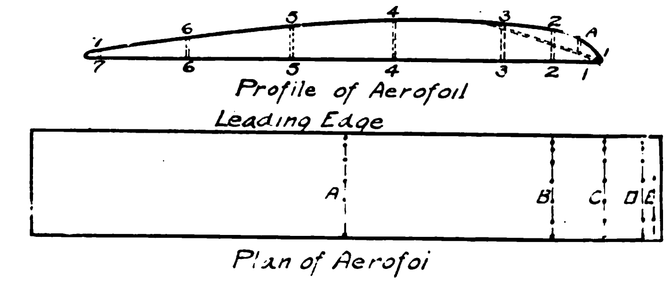


FIG. 5. SECTION EMPLOYED AT THE N. P. L. IN OBSERVING PRESSURE DISTRIBUTION ON AN ENTIRE WING.

tions parallel to the median section. The actual methods were similar to those already described in considering pressure distribution over a median section, and the same remark applies to the resolution of components normal and perpendicular to the chord and their subsequent summation. The centers of pressure for each section were obtained by taking moments by a process of elementary mechanics, a similar process is fully described in the *Bulletin de l'Institut Aérotechnique.* Normal forces were again taken as a measure of the lift, and forces parellel to the chord as a measure of the drag, skin friction being neglected. The resolution of forces along and perpendicular to the chord, instead of along and per-

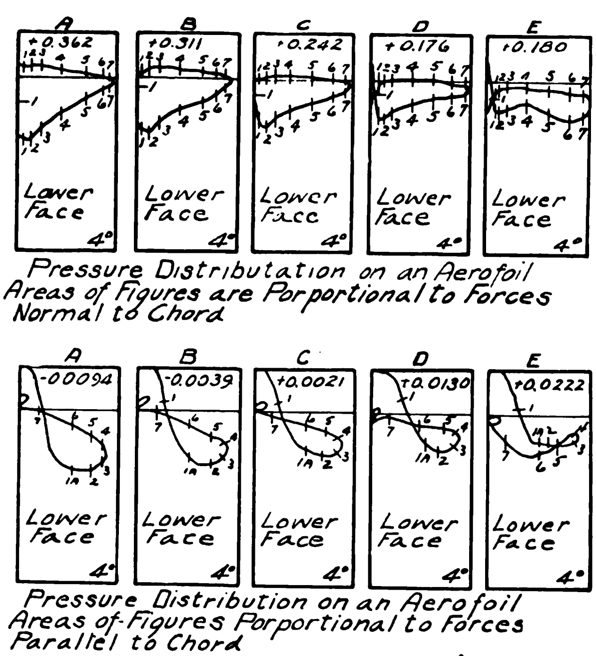


FIG. 6.

pendicular to the relative wind, involved the error already mentioned, unimportant, however, except at large angles.

In Fig. 6 are shown the curves of normal and parallel forces at the five sections for various angles of incidence. In Fig. 7, the contributory effects of each section of the wing is clearly illustrated by curves giving lift, *L/D* and center of pressure for each section. The following observations can be made from this data:

(a) As each section, beginning with the median, is considered, the distribution on the upper surface from high

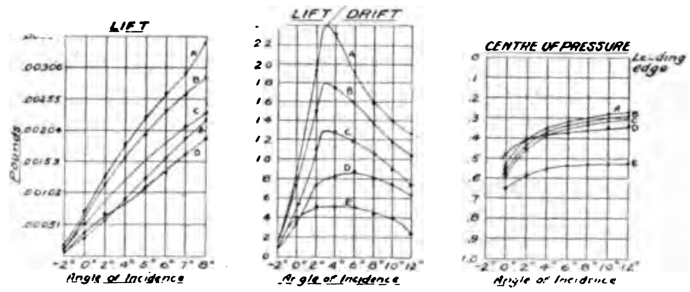


FIG. 7. CONTRIBUTORY EFFECTS ON VARIOUS ELEMENTS OF A WING AT VARIOUS ANGLES OF INCIDENCE. SECTIONS LOCATED AS IN FIG. 5.

suction forward and low suction aft alters progressively until when the tip is reached the highest suction occurs in the neighborhood of the trailing edge.

(b) On the lower surface, the positive pressures found over the central portion of the wing fall off, and eventually change sign, so that near the tip almost the whole section is under suction.

(c) At the same time, the areas of the curves of normal forces on upper and lower surface decrease at first to about $^2/_3$ of their original value, but subsequently increase as the tip is reached.

(d) Again as we move from the median section outward, the areas of diagrams proportional to parallel forces change from negative values (which oppose drag) to appreciable positive values, so that drag of the sections increases very rapidly in the neighborhood of the wing tips. Thus maximum *L/D* at *A* is 24, at *E* it is only 5.

(e) It is the variations in pressure distribution as we move out laterally which cause the center of pressure of the whole wing to move back.

This seems to demonstrate clearly that at the sides of the wing section there is a considerable amount of lateral flow, which prevents the establishment of a régime as efficient as at the center, where the air does not escape but follows the contour of the wing.

It is now also clear why increased aspect ratio is advantageous: As aspect ratio is increased, the inefficient action of the exterior sections assumes less importance. Without further research it is, however, impossible to say whether increase in aspect ratio leaves the aerodynamical conditions at the median sections unaltered, or whether it improves conditions everywhere on the wing except on the lateral tip.

These experiments may not be of immediate application in design, but may serve to give a better conception of what may be expected when a wing is varied in plan form. Besides the effects of varying ratio, these considerations would tend to explain the effects of raking.

Distribution of Pressure Over Entire Surface of Wing and Curves of Equi-Pressure

Eiffel employs an instructive method of curves of equipressure over the entire surface of the wing. He has obtained such curves for flat plates and for cambered surfaces, but has unfortunately not carried his analysis very far, and gives us nothing beyond a graphical idea of the actual distribution of pressures. In Fig. 8 we find curves of equipressure and pressures at various sections for the Nieuport wing, which are somewhat more suggestive.

In the Nieuport wing, the sections, while preserving their general character, thin down as we move from the median section outward. This has the effect of maintaining nearly the same character of pressure distribution on all sections.

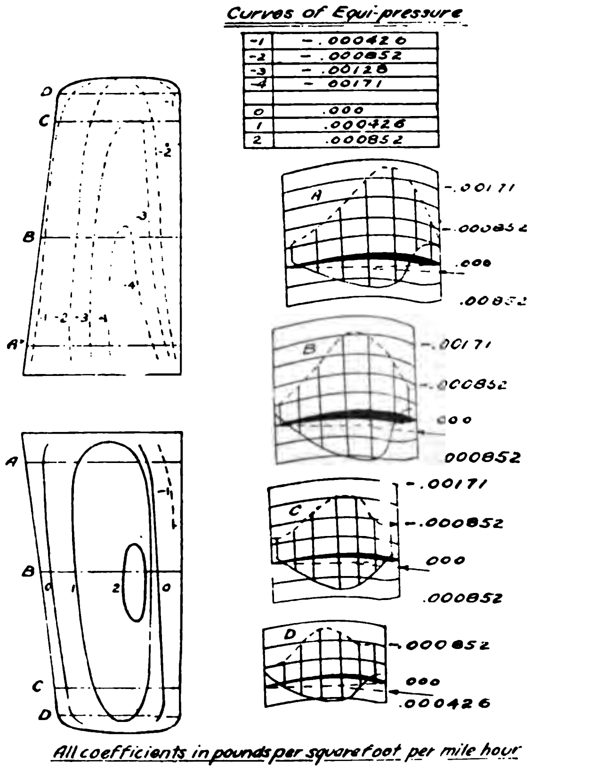


FIG. 8. DISTRIBUTION OF PRESSURE ON NIEUPORT WING.

The outer sections have smaller values, it is true, but the maximum suction on them is still not far from the leading edge. This from considerations of preceding paragraphs tends to minimize the aerodynamic inefficiency on the outer section. A wing such as the Nieuport wing might, therefore, be very valuable, but further experiments would be necessary before this point could be definitely settled.

Relative Importance and Interdependence of Two Surfaces

In Fig. 9 are shown curves due to the National Physical Laboratory, which show the distribution of pressures along and normal to the chord on the upper and lower surfaces of two wing sections. The sections are alike in their upper surfaces, but one of them, section 2, is hollowed out, while the other, section 1, has a plane under surface.

As we have already stated, the sum of two forces normal to the chord is scarcely distinguishable from the lift. And it can be readily seen from these curves that in both sections the upper surface contributes all the normal force at 2 degrees and nearly three-quarters of it at 12 degrees.

It can be deduced from this that the lower surface of a wing section provides not more than one-quarter of the total lift. We notice further that the curves for the upper surfaces in both sections are practically identical.

Section 1 has no components parallel to the chord; in section 2, as can be seen from the curves, the lower surface contributes very little of such components. Up to 7 degrees, the lower surface gives an upward force which helps to diminish drag; above this it has down a "downward" and detrimental effect. It can be seen here that the effect of the upper surface is similar in both sections.

From these considerations we can apparently conclude:

(a) That it is the upper surface of a wing which is by far the most important.

(b) That hollowing out a section has very little effect either on its lifting power or on its efficiency.

A somewhat similar conclusion was arrived at by Bendemann in examining a heavily cambered almost circular propeller section; the camber of the lower surface scarcely affected the value of the coefficients.

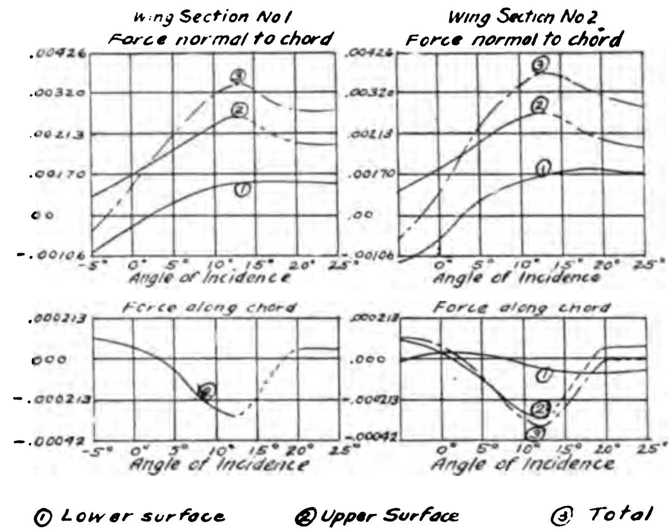


FIG. 9. DIAGRAMS ILLUSTRATING THE INTERDEPENDENCE AND RELATIVE IMPORTANCE OF TWO SURFACES OF A WING SECTION. COEFFICIENTS ARE IN FOOT POUNDS PER SQUARE FOOT PER MILE HOUR.

Distribution of Pressure; the Principle of the Dipping Front Edge. Why a Wing Section Is Advantageous as Compared with a Flat Plate

We know from the pressure diagrams that for any wing such as that in Fig. 10, the pressure at A is negative, while

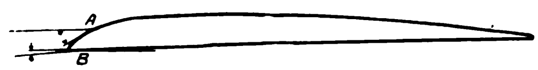


FIG. 10. DIAGRAM ILLUSTRATING THE PRINCIPLE OF THE DIPPING FRONT EDGE.

that at B is positive. This seems paradoxical, since A would appear to be facing the wind, while B is sheltered from it. Fage gives a very good explanation of this phenomenon commonly known as the "Principles of the Dipping Edge." Photographs, such as we have already given in Chapter 3, show that the wind is deflected upward as it approaches the leading edge of the wing. A, although it faces the general wind direction, is thus screened from it, and becomes a region of low pressure; while on B the relative wind impinges directly and receives a slight downward deflection.

From somewhat similar considerations, we are now in a position to explain roughly why a wing section is so much more advantageous than a flat plate:

(1) The suction on the upper surface of a wing toward the trailing edge is much greater than that for a flat plate, explainable by the principle of the dipping front edge. And a greater suction implies a greater lifting power.

(2) From the pressure distribution curves on the median section, it can be seen at once that the greater part of the force on the upper surface is due to the suction in the region of the leading edge.

Hence in summing up components parallel to the chord, for a good wing, the resultant force will tend to be "up-wind," and tend to nullify the skin friction, reducing the total drag. In a flat plate no such advantageous action will be present. A wing will, therefore, give a greater lifting power and a bigger lift to drag ratio.

References for Part I, Chapter 7

INVESTIGATION OF THE PRESSURE IN A MEDIAN SECTION OVER THE UPPER AND LOWER SURFACES OF THREE AEROFOILS

British Report, 1911-1912. No. 60. Page 62.

INVESTIGATION OF THE DISTRIBUTION OF PRESSURE OVER THE ENTIRE SURFACE OF AN AEROFOIL

British Report, 1912-1913. No. 73.

EFFECT OF VARIATION OF SPEED AND SIZE, LIFT AND DRAG COEFFICIENTS

British Report, 1912-1913. No. 72. Page 81.

DISTRIBUTION OF PRESSURE AT MEDIAN CROSS SECTION OF VARIOUS SURFACES

"The Resistance of Air and Aviation," Eiffel (trans. Hunsaker), page 70.

DISTRIBUTION OF PRESSURE

Bulletin de l'Institut Aérotechnique de l'Université de Paris, 1913, Etude des Surface au Chariot Eléctrique.

"Curves of Equi-Pressure for the Nieuport Wing," Eiffel (Hunsaker), page 171.

INTERDEPENDENCE OF THE TWO SURFACES OF A WING

British Report, 1911-1912. No. 60. Page 68.

"Luftschrauben-Untersuchungen," F. Bendemann. *Zeitschrift für Flugtechnik*, 1911-1912.

PRINCIPLE OF THE DIPPING FRONT EDGE

A. Page. "The Aeroplane," page 17

Chapter VIII
Biplane Combinations

Monoplane surfaces are aerodynamically the most efficient. Biplane combinations of any kind introduce interference between the planes, a diminution of the suction on the lower plane, with a consequently diminished efficiency. But as airplanes increase in size the difficulties of suitably bracing monoplane surfaces become very great, and their lifting capacity inadequate, and biplane construction must be resorted to.

Another important aspect of biplane construction is the possibility of obtaining longitudinally stable arrangements by staggering or displacing the wings relative to one another, and by introducing small angles between their planes, which is known as décalage.

The effects of staggering the planes for convenience of construction or with a view to increasing the range of view are to be considered within the province of practical construction.

Orthogonal Biplane Arrangements with Varying Gap Between Planes

An orthogonal biplane, as shown in Fig. 1, Setting No. 1, is one in which the lines joining the leading and trailing edges of the two wings are both at right angles to the chord.

Experiments to which reference is given at the end of this section to determine the aerodynamic coefficients of such combinations with varying gap between planes have been carried

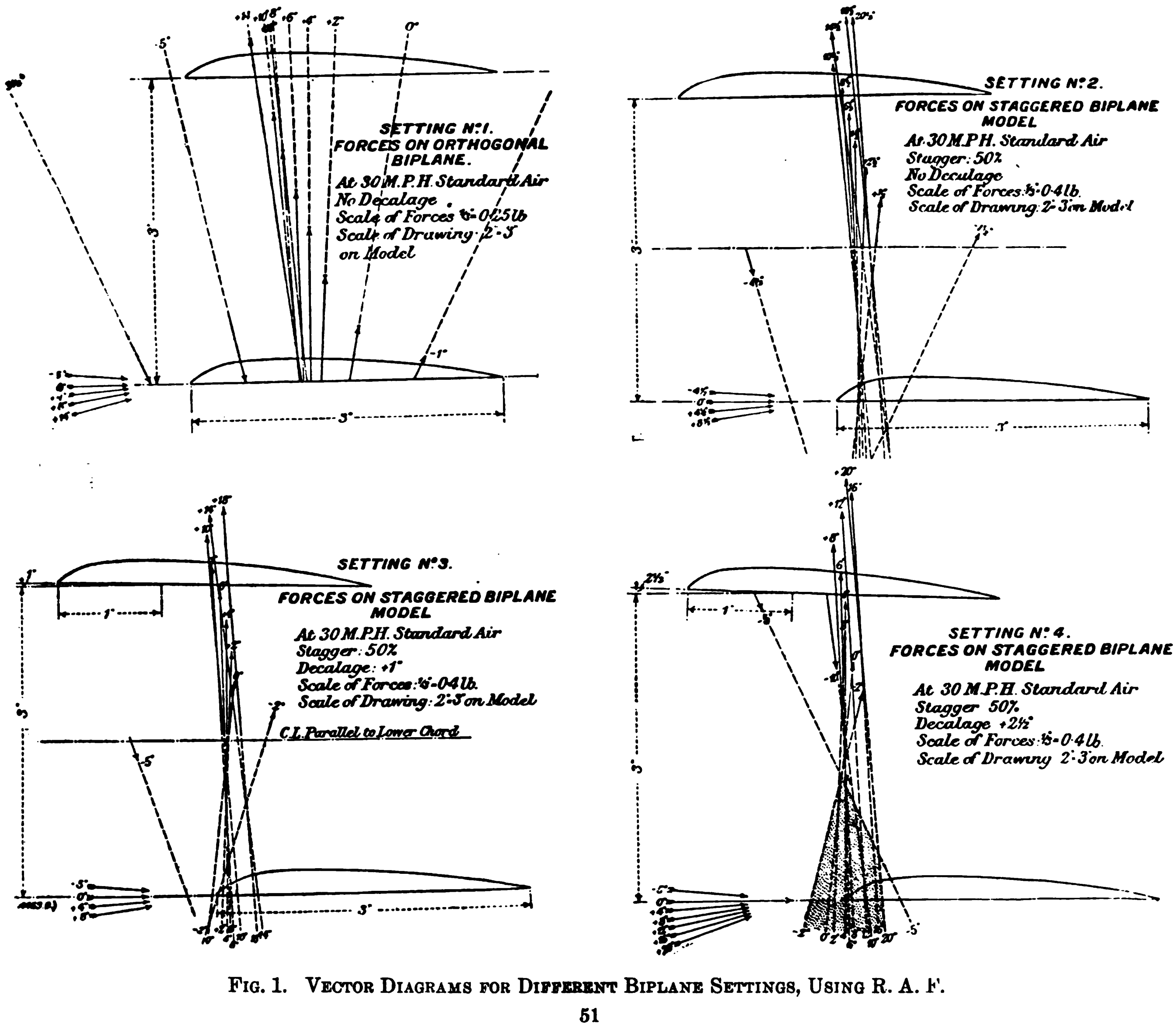


FIG. 1. VECTOR DIAGRAMS FOR DIFFERENT BIPLANE SETTINGS, USING R. A. F.

out solely on wings of an antiquated type, and it is by no means certain that similar values would apply exactly to modern wings. In default of further exhaustive experimentation, the N. P. L. values must be taken as a guide, however. The results of the N. P. L. experiments showed that for normal angles of incidence:

(1) Drag per unit area of biplane combination was not appreciably greater than that of a similar monoplane surface.

(2) The lift coefficients as compared with a monoplane surface decreased considerably, and that the loss was the greater the smaller the ratio between gap and chord.

(3) Loss in value of lift/drag follows:

On the basis of these experiments the following table can be employed:

TABLE 1.

FOR ANGLES OF INCIDENCE IN NORMAL FLIGHT

Ratio of gap to chord	0.40	0.80	1.00	1.20	1.60
Factor for K_y to reduce biplane lift from coefficients of a monoplane surface	0.61	0.76	0.81	0.86	0.89
Factor for K_y/K_x to reduce from coefficients of a monoplane surface	0.75	0.79	0.81	0.84	0.88

Distribution of Forces Between the Upper and Lower Wings of a Biplane

By an indirect deduction from Dr. Hunsaker's experiments on the triplane the following figures may be given for the distribution of lift on the upper and lower wing of a biplane, with ratio gap to chord 1.2:

TABLE 2.

Angle of Incidence	Percentage Lift Upper Wing	Percentage Lift Lower Wing
0	62%	38%
2	55%	45%
4	54%	46%
8	53%	47%
12	54%	46%

It is possible that the upper wing does not only carry a greater proportion of the lift, but that it also has a better L/D ratio and has a proportionately small drag. Still the standard assumption, as used by Dr. Zahm, among other eminent authorities, that 55 per cent of all the forces acting on a biplane may be taken as acting on the upper plane is sufficiently accurate for all practical purposes. The distribution of forces between the two planes is only useful in stress calculations in design, where an error of a few per cent will have little or no importance. In Eiffel's earlier experiments some interesting data for pressure distribution on the upper and lower wings of a biplane are given which bear out the above values.

Distinction Between Static and Dynamic Stability

It is important at this stage of the work to draw a distinction between static and dynamic stability. An airplane with static longitudinal stability has a righting moment

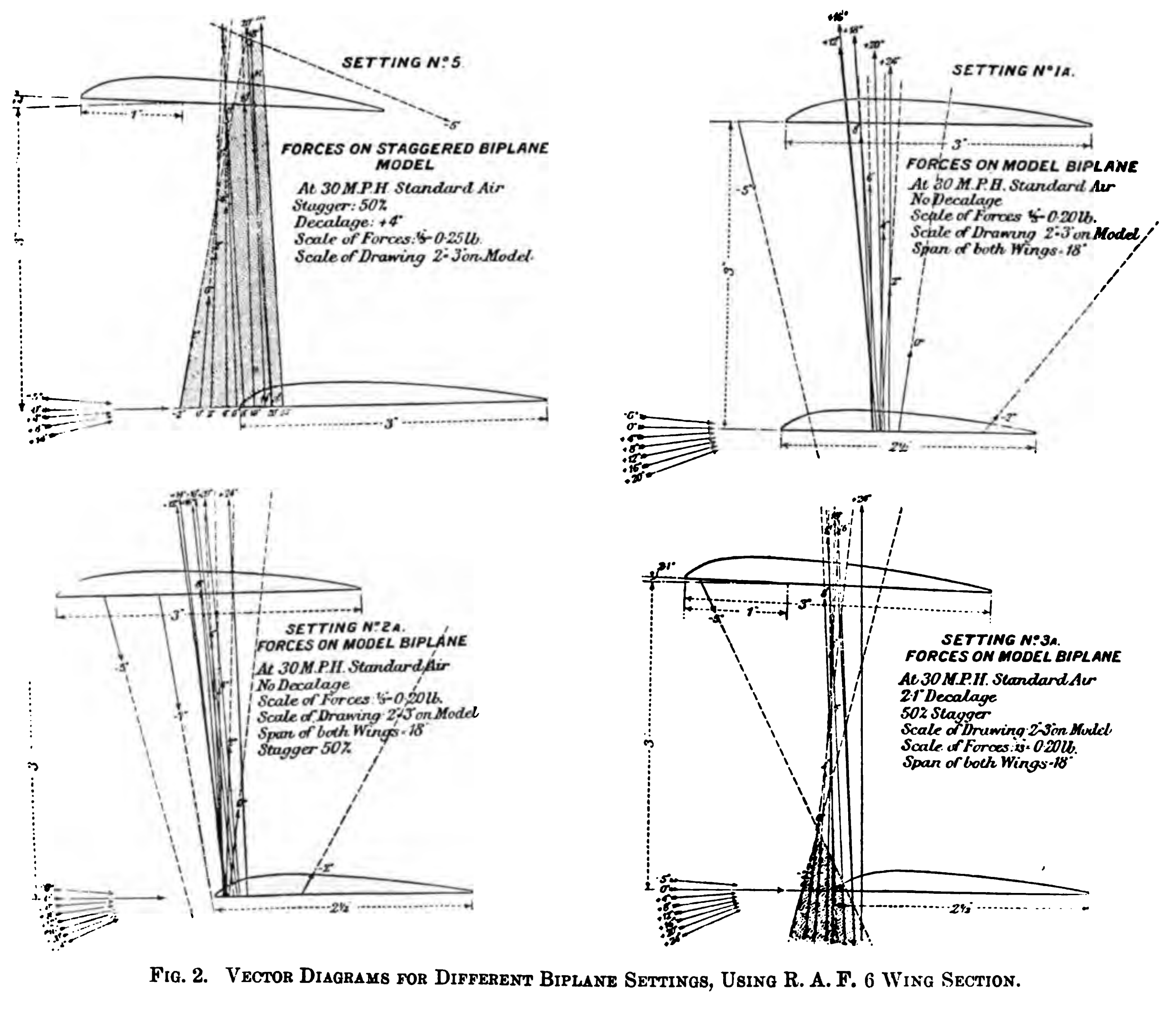


FIG. 2. VECTOR DIAGRAMS FOR DIFFERENT BIPLANE SETTINGS, USING R. A. F. 6 WING SECTION.

when displaced from its position of equilibrium, which tends to bring it back to the position of equilibrium. This righting moment may be so violent, however, that the airplane may acquire a considerable rotational velocity (pitching velocity),

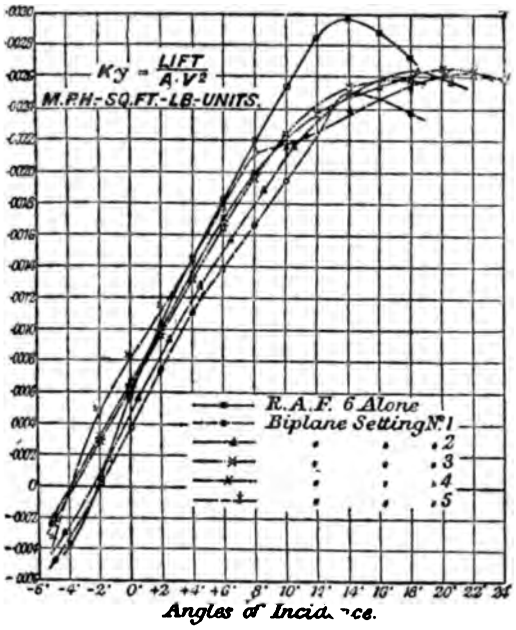


FIG. 3. LIFT AND DRAG COEFFICIENTS.

overshoot its position of equilibrium, and then, with the intervention of a righting moment in the opposite direction, oscillate back and forth. In fact, the greater the static stability, the more violent may be the longitudinal oscillations.

In addition, therefore, there must be dynamic stability supplied by large tail surfaces, with a long arm about the center of gravity to damp out the oscillations which the static stability alone is unable to subdue. A concise but authoritative

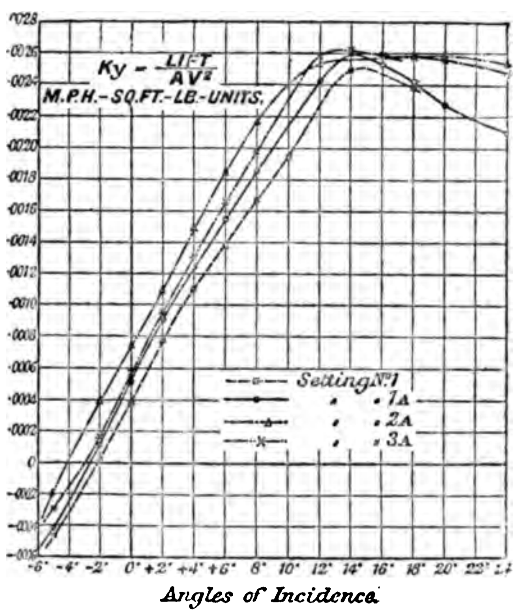


FIG. 4. LIFT AND DRAG COEFFICIENTS.

discussion of dynamic stability has appeared in AVIATION AND AERONAUTICAL ENGINEERING (see appended references).

Stable Biplane Arrangements

We have seen that it is possible to secure a large degree of static stability at the expense of some loss in efficiency by the employment of wings with reversed curvature at the trailing edge. It is possible to insure static stability also by the employment of biplane combinations with stagger and décalage. Dynamic stability without preliminary static stability is im-

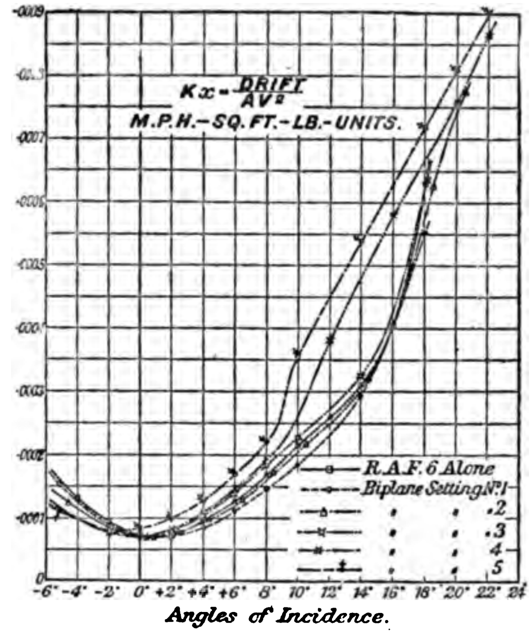


FIG. 5. LIFT AND DRAG COEFFICIENTS.

possible, but if an airplane is statically stable, dynamic stability is certainly possible.

Dr. Hunsaker investigated a great number of biplane arrangements at the Massachusetts Institute of Technology, with

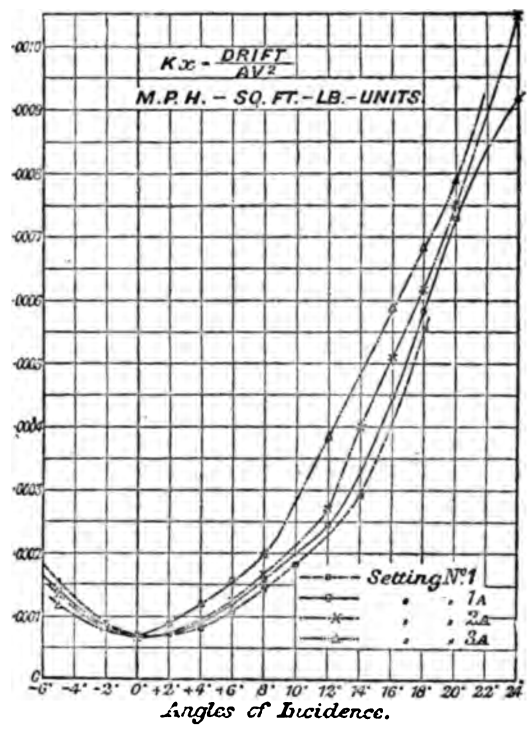


FIG. 6. LIFT AND DRAG COEFFICIENTS.

varying degrees of stagger and décalage, and found that with certain combinations:

(1) Static longitudinal stability could be obtained with but little loss in aerodynamic efficiency.

(2) By suitable arrangements, the lift curve at the burble point can be flattened out and made to maintain its maximum for a wide range. This is particularly valuable, because it

eliminates the danger usually attending stalling altitudes. With a sharp drop in lift at the burble point, the loss in sustentation beyond a certain angle may be so great that the machine may drop.

Results of Experiments on Biplanes with Stagger and Décalage

In Table 3 are given the summarized results for a series of tests on such combinations. In Figs. 1 and 2 are shown the corresponding combinations with the vector diagrams; in Figs. 3, 4, 5 and 6 are shown K_y and K_x curves, and in Fig. 7 are plotted these K_y against K_x curves for all the settings.

To judge of the stability of any combination it is necessary to assume a number of positions for the center of gravity, to assume a normal flying angle of incidence, and to see whether displacement from the normal flying position is followed by the correct righting moment about the center of gravity. If, for instance, the center of gravity for the setting No. 4 of Fig. 1 is placed as shown, between the vectors for 4 degrees and 6 degrees incidence, with normal flying angle 5 degrees, there will be statical stability. If the airplane dives to 2 degrees, the resultant force will have a clockwise moment about the center of gravity and will tend to right the machine. If the airplane stalls to 8 degrees, the resultant force will have a counter-clockwise moment and will again tend to restore the biplane to its normal position.

In Table 3 the various settings are classified as stable and unstable, and it forms a very useful exercise to examine each combination from this point of view. The comparative values aerodynamically and the lift at the burble point are brought out clearly by Table 3 and by the K_y and K_x curves.

Even with these extensive tests it is impossible to draw definite conclusions as to the selection of any particular type, and the results should be regarded as more qualitative than quantitive. The qualitative results would prevent any fantastic combination being employed.

Some of the main conclusions may be summarized as follows:

(1) Stagger alone improves the aerodynamical qualities of a biplane, and flattens out the burble point, moves the vectors of force forward, but does not increase the stability to any appreciable extent.

(2) Cutting down the lower wing of a biplane does not improve the stability, but it lessens interference, improves the aerodynamic efficiency, and flattens out the burble point.

(3) Increasing décalage combined with stagger produces progressive stability, but at the expense of aerodynamic efficiency.

(4) Among the most promising arrangements seem to be:

No. 4. Décalage 2½ degrees, stagger 50 per cent. The stability is gained at the expense of but 4 per cent of the maximum lift/drag ratio, while a gain is obtained in all other properties.

No. 3A. Décalage 2.1 degrees, stagger 50 per cent, lower chord 83 per cent of the upper chord. Here the stability is also attained at a loss of but 4 per cent on maximum lift/drag ratio, while the lift curve remains at its maximum over a range of 12 degrees.

Comparison of Aerodynamic Losses Involved in Obtaining Stability by Reversed Curvature Wings and by Stagger—Décalage Combinations

For reverse curvature wings giving static longitudinal stability the ma imum lift is 17 per cent less and the maximum lift/drag ratio is about 14 per cent less than for a simple orthogonal biplane, as seen from the last column of Table 3.

TABLE 3.

CHIEF PARTICULARS FOR STABLE BIPLANE ARRANGEMENTS

Type	Gap	Stagger	Upper Chord	Lower Chord	Decalage Degrees	Max. K_y/K_x	Max. K_y	K_y/K_x where K_y = 0.0005	K_y/K_x where K_y = 0.0018	Range of Flat Burble Point in Degrees	Remarks on Stability with reference to Figs. 1 and 2
Monoplane......	..	..	C	..	..	1.15	1.16	0.90	1.24	2°	Unstable.
Biplane No. 1...	C	0.	C	C	0.0	1.00	1.00	1.00	1.00	2°	Unstable.
Biplane No. 2...	C	0.50C	C	C	0.0	1.00	1.06	1.00	1.02	2°	Resultant forces from 2½° to 10½° intersect near a single point. If this point be the center of gravity there will be no pitching moment throughout this range. For the extreme range of flying angles from 1½° to 20½° the equilibrium is stable.
Biplane No. 3...	C	0.50C	C	C	1.0	0.95	1.03	1.00	1.01	6°	The force vectors for angles from 0° to 10° intersect near a point. If the center of gravity be at this point the equilibrium is neutral from 0° to 10°, stable from 10° to 18° and unstable from 0° to −5°. If center of gravity be placed low, at about the intersection of the vector for 4° and the lower chord, the equilibrium is stable for all the range from −2° to +18°.
Biplane No. 4...	C	0.50C	C	C	2.5	0.95	1.03	1.00	1.02	4°	For a center of gravity located anywhere in the lower triangle bounded by the vectors for −2° and −5° the equilibrium is stable longitudinally throughout the entire range of pitching angle −5° to +20°. Very good arrangement.
Biplane No. 5...	C	0.50C	C	C	4.0	0.87	1.04	0.90	0.99	4°	Excessive stability. Machines suitable for amateurs.
Biplane No. 1A.	C	0.	C	0.83C	0.0	1.04	1.04	0.90	1.05	4°	Longitudinally unstable.
Biplane No. 2A.	C	0.50C	C	0.83C	0.0	1.04	1.04	0.93	1.08	12°	Unstable.
Biplane No. 3A.	C	0.50C	C	0.83C	2.1	0.96	1.03	0.96	1.05	12°	Longitudinal stability for any center of gravity located within the lower triangle formed by the vectors for −2° and −5°. This will be the case for a heavy sea-plane.
Reverse Curvature Wings...	C	0.	C	0.	0.0	0.86	0.83	1.21	0.88	2°	Longitudinally stable.

C = chord length (upper).

K_y/K_x and other aerodynamic coefficients referred to the othogonal standard biplane as unity.

With a stagger décalage combination there is an actual increase in the maximum lift, while the L/D loss is only 4 per cent. The constructional difficulties in the region of the rear spar are also avoided. On the other hand, stagger involves the biplane drag was greater than that of the monoplane, while at high angles the biplane gave the better qualities. The later Massachusetts Institute of Technology experiments gave diametrically opposite indications. Since these experiments

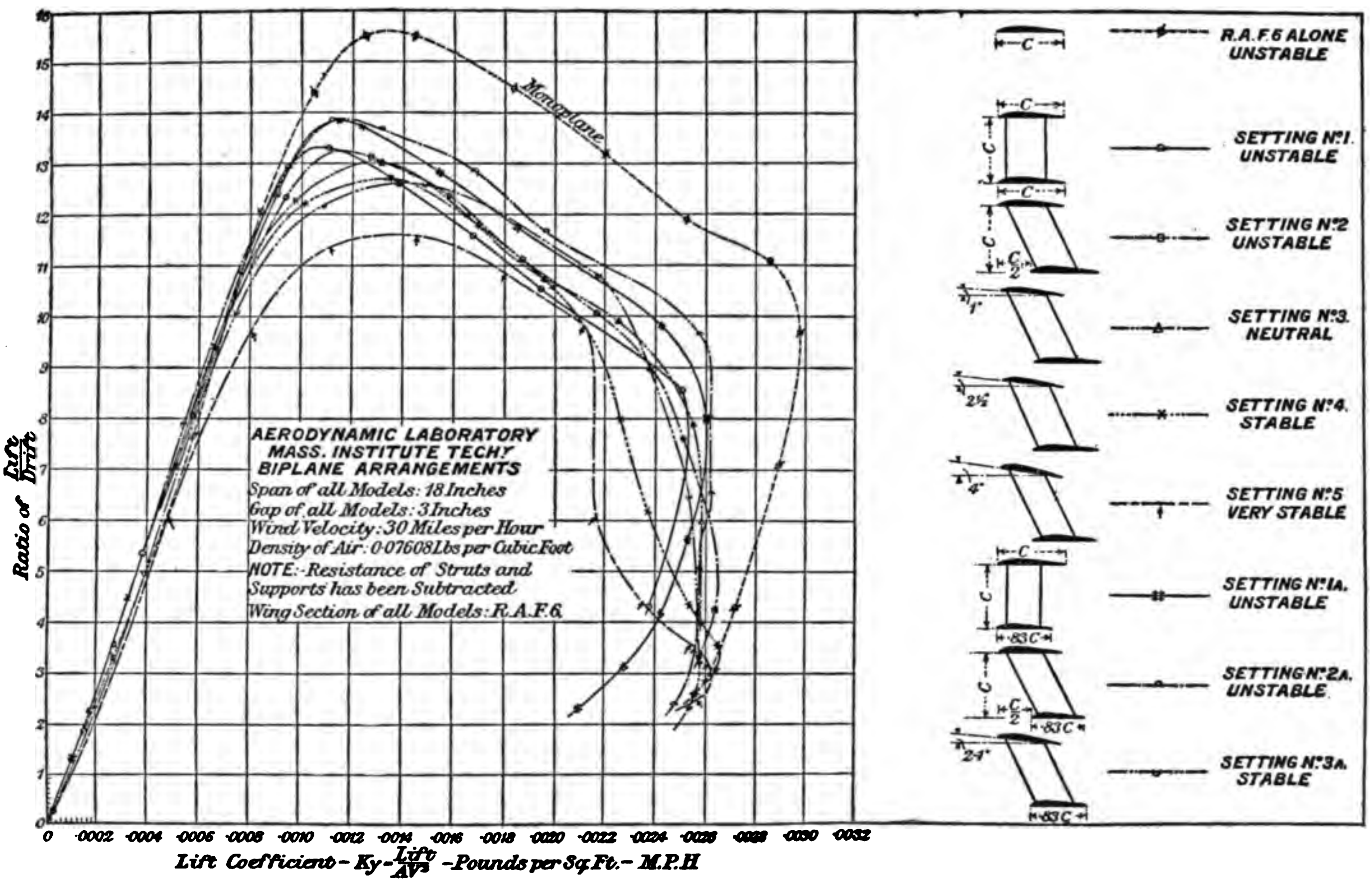


FIG. 7. LIFT COEFFICIENTS PLOTTED AGAINST DRAG FOR VARIOUS SETTINGS.

increased length and resistance of wing struts and increased stresses in the drift bracing of the wings.

The relative merits of the two systems can only be decided upon by a practical comparative experience of the two types. In the authors' opinion, the stagger-décalage system is more likely to give good results than the reversed curvature wing system for ordinary machines. For very high speed machines, flying at a small angle of incidence, however, the reversed curvature biplane offers 20 per cent less resistance than the orthogonal biplane with R. A. F. 6 wing section. In such machines, where a low maximum K_y coefficient and high landing speeds are permissible, the reversed curvature wing might be very advantageous from the point of view of high maximum speeds.

Aerodynamic Comparison Between the Monoplane and the Biplane

In Table 1 are given the correcting factors from monoplane values for biplanes with varying gap/chord ratios from the N. P. L. experiments. These will, although based on a wing section of an antiquated type, as already mentioned, be quite correct enough for angles of normal flight, 4, 6 or 8 degrees incidence. But for very low angles of incidence and for very high angles of incidence there is a discrepancy between the results obtained by the British investigators and by Dr. Hunsaker. The former concluded that at low angles of incidence were conducted at a later date, and were carried out with R. A. F. 6 wing sections, they are probably worthy of more credence. The following table has been deduced from the curves of Fig. 7, where K_y is plotted against K_x:

TABLE 4.

LIFT/DRAG RATIO AND K_x FOR ORTHOGONAL BIPLANE, R. A. F. 6 WING SECTION, GAP/CHORD RATIO 1, GIVEN AS PERCENTAGE OF MONOPLANE VALUES FOR THE SAME K_y

K_y	K_y/K_x	K_x
0.0004	110	90
0.0006	107	93
0.0008	99	101
0.0012	85	115
0.0016	85	115
0.0020	75	125
0.0024	73	127

To consider L/D and K_x for the same values of K_y for monoplane and biplane is really a much fairer comparison than to consider L/D and K_x for the same angles of incidence. It really matters very little what the angles of incidence for biplane and monoplane are, provided we have the same K_y and the same sustaining power at the same speed.

From Table 4 one would conclude that the biplane has a very distinct advantage for a high-speed scout. Apparently at a high speed, and hence a low lift coefficient, the biplane resistance is 10 per cent less than the monoplane resistance. This is an appreciable saving. For a machine which must fly slowly, and consequently with a high lift coefficient, the biplane resistance is from 15 to 25 per cent greater than the monoplane resistance.

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Chapter IX

Triplane Combinations — Uses of Negative Tail Surfaces

In an article on " The Aerodynamical Properties of the Triplane," by J. C. Hunsaker and T. H. Huff, published in the November 1, 1916, issue of AVIATION AND AERONAUTICAL ENGINEERING, the reader will find a complete treatment of the aerodynamic properties of the triplane, with a complete record of the experimental results obtained at the Massachusetts Institute of Technology. It remains for us only to summarize the main results, and to review recent constructional applications of the triplane principle.

The main conclusions from these experiments are:

(I.) At the stalling angles such as 16 degrees the triplane and biplane give nearly the same maximum lift; the triplane has a materially lower resistance at this angle, giving a much better performance at slow speed. Thus the L/D ratio at 16 degrees is 4.5 for the monoplane, 5.6 for the biplane, and 6.5 for the triplane.

(II.) At angles below 12 degrees the drag coefficient is not greatly different in the three cases, but the lift for the triplane is considerably reduced; it is inferior to that of the biplane which again is inferior to that of the monoplane.

(III.) The best L/D for the triplane combination is only 12.8 as compared with 13.8 for the biplane, and 17 for the monoplane.

(IV.) The center of pressure motion is almost identical with that of the biplane. We have seen previously that the center of pressure motion for the biplane is nearly that of the monoplane. This demonstrates that the commonly made assumption of monoplane center of pressure motion for a wing of a biplane also holds for the triplane. This is an important fact in view of the methods employed in stress diagrams.

The experimental results for K_y and L/D for the triplane as compared with the monoplane and biplane are summarized in Table 1:

TABLE 1

TABLE SUMMARIZING COMPARATIVE VALUES OF K AND L/D FOR MONOPLANE, BIPLANE AND TRIPLANE.

	MONOPLANE.		BIPLANE.		TRIPLANE.	
i	Actual K_y	Percentage.	Actual K_y	Percent. of Monoplane.	Actual K_y	Percent. of Monoplane.
0	.000486	100	.000432	88.8	.000404	83.0
2	.00103	100	.000864	83.8	.000776	75.4
4	.00145	100	.00123	85.4	.00109	75.7
8	.00218	100	.00186	85.2	.00169	77.4
12	.00278	100	.00244	87.6	.00226	81.2
16	.00277	100	.00273	98.5	.00267	96.4
	L/D	L/D	L/D	L/D	L/D	L/D
0	8.6	100	6.3	73.2	6.1	70.8
2	16.3	100	12.2	74.7	11.4	69.8
4	16.8	100	13.8	82.0	12.8	76.1
8	13.8	100	11.3	81.9	11.1	80.4
12	10.0	100	9.5	95.0	8.9	89.0
16	4.5	100	5.6	124.0	6.5	145.0

Interference in Triplanes

Dr. Hunsaker's paper also deals fully with interference in triplanes. It is important in the structural design of the wing girder to know what portion of the lift and drag to attribute to each wing. The comparitive efficiency of each wing is also important from the point of view of overhang. It appears from Table 2 that the upper wing is very much the most effective of the three and the middle wing the least effective. The very poor lift of the middle wing is caused by the interference with the free flow of air due to the presence of the upper and lower wings.

One interesting point brought out by Dr. Hunsaker toward the end of his paper was the fact that when the effects of the upper and lower wings were combined, results identical with that of a simple biplane combination were obtained. This would tend to show that the interference in the case of a triplane is similar to interference in the case of a biplane. The upper wing of a triplane would seem to be influenced by the middle wing in the same way that the upper wing of a biplane is influenced by the lower wing of a biplane. Again the lower wing of a triplane would seem to be influenced by the middle wing in the same way that the lower wing of a biplane is influenced by the upper wing. These are important considerations to be kept in mind when modifications of the triplane are attempted such as stagger, overhang, décalage, etc.

TABLE 2

TABLE OF VALUES FOR LIFT AND L/D FOR EACH WING OF A TRIPLANE COMBINATION AS RATIOS TO LIFT AND L/D OF MIDDLE WING.

Angle of Incidence.	Lift Upper.	Lift Middle.	Lift Lower.	L/D Upper.	L/D Middle.	L/D Lower.
0	2.68	1.0	1.82	3.63	1.0	2.30
2	2.14	1.0	1.76	3.18	1.0	2.13
4	1.91	1.0	1.64	2.59	1.0	1.69
8	1.56	1.0	1.36	1.49	1.0	1.37
12	1.56	1.0	1.31	1.30	1.0	1.34
16	1.49	1.0	1.20	1.22	1.0	1.17

Some Considerations for Triplanes

There are two types of airplanes, quite dissimilar, for which triplanes have been employed in this country, the huge Curtiss flying boats, and the recent Curtiss speed scout. It is interesting to consider what the possible advantages of the triplane are at these two extremes of design.

In the heavy types, particularly in seaplanes, the increased size must be developed without increase in landing speed. To insure about the same landing speed, the loading must remain at a figure of about 5 pounds per square foot. And for an aeroplane of four times the ordinary weight the wing area must be increased in like proportion. Monoplane construction is obviously impractical for such great areas of wings, and even with the biplane there is an enormous wing span. Such a span introduces great difficulties from the stress point of view and from the point of view of housing and handling. The employment of a triplane enables the span to be kept within reasonable dimensions and also permits the employment of larger aspect ratios.

At high angles the lift of a triplane is only 1.1 per cent less than that of a biplane of the same area. At 16 degrees the L/D ratio of a triplane is 16 per cent better than that of a biplane. At stalling attitudes, the triplane has therefore very decided advantages, giving a greater reserve power at low speeds in alighting. At 4 degrees incidence for best L/D the triplane does not show up so well, and requires an increase in power of about 6 per cent. It would seem as if the 6 per cent increase in power can be compensated for by the possibility of a less heavy type of construction with decreased span. Also there is the possibility of employing much greater aspect ratios than in biplane work, and this may compensate to some extent for the losses due to extra interference.

Triplanes for Fast Speed Scouts

From the photographs issued by the Curtiss Company, it seems clear that the original machine (see AVIATION August 15, 1916) was transformed (see AVIATION October 15, 1916) by placing the triplane wing structure on the same body structure as for the biplane. With approximately the same wing area divided between these planes the aspect ratio was increased very considerably, without exaggerating the span, giving some aerodynamical advantage. The extremely narrow blade like wing permitted the single plane bracing system to be used with much greater security. The employment of a single plane bracing system cuts down resistance very considerably, and if this bracing system is only possible with triplane narrow blade construction, then triplane construction would be a very sound tendency in the design of small fast machines.

Use of Negative Tail Surfaces

In Chapter 6 we saw the possibility of using a negative tail surface so as to give static longitudinal stability, and in the problem which follows a definite case will be taken as an illustration of this possibility. In other methods of attaining stability such as employment of the reversed curvature wing or of stagger-décalage combinations, dynamic stability might not follow the static stability unless tail surfaces were employed. In such cases the tail surfaces would probably have to be placed at zero angle to the wings, or even at a small positive angle. It is for this reason that the stable biplane arrangements, apparently so advantageous, are not more frequently used in practice.

Effect of Influence of the Wash of the Wings on Stabilizer Surface

Eiffel in his later experiments conducted some tests on tandem wings. One important result of these tests was to prove that an airplane built with tandem wings would be

FIG. 1. DIAGRAM FROM EIFFEL TO SHOW DEVIATION OF STREAM-LINES BY "DOWN-WASH" OF WINGS.

aerodynamically disadvantageous. Another result was that the down wash of the front wing would change the flow relative to the rear wing so that the angle of incidence of the latter would be smaller than that of the former. The allowances made for the change in the angle of incidence were arrived at indirectly by measuring lift and drag on the second wing while in the presence of the first, and are not entirely reliable. Fig. 1 represents Eiffel's conclusions diagramatically for a specific case.

The relative wind for the front wing is horizontal; the chord of the front wing is at 10 degrees to the horizontal, the chord of the rear wing is at 4 degrees to the horizontal, with a décalage of 6 degrees. Owing to the downwash of the front wing the stream lines at the rear wing now make a negative angle of 7 degrees with the horizontal, and the rear wing is actually at a negative angle of 3 degrees to the stream.

Owing to the fact that the tail surfaces of a machine are so much further away from the wings, the deviation will be less perceptible and the following practical formula gives satisfactory results:

If i = angle of incidence of the wings,

Deviation of stream = $(\frac{1}{2}\, i + 1)$

To take a concrete case, if the wings are at 8 degrees incidence, and the stabilizer is placed at a negative angle of 2 degrees to the chord, then its incidence will be

$$\begin{matrix}(8-2) & - & (\frac{1}{2}\, i + 1) \\ (8-2) & - & (8/2 + 1)\end{matrix} = 1$$

This deviation has an important bearing in the design of stabilizing surfaces on the angle of setting between the chord of the wing and the line of the stabilizing plane.

Problem on the Design of Tail Surfaces to Give Longitudinal Static Stability

The requirements of static longitudinal stability may be briefly stated as follows:

(1) The machine must be in stable equilibrium at some angle of incidence, generally the angle of normal flight, say 6 degrees for purposes of illustration.

(2) If the angle of incidence of the aeroplane is from any cause less than 6 degrees, there should be a positive restoring

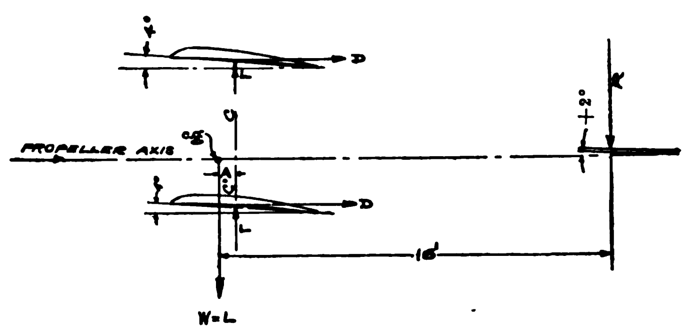


FIG. 2. DIAGRAM ILLUSTRATING STABILIZER PROBLEM.

moment or stalling moment; if for any reason, greater than 6 degrees, there should be a negative or diving moment.

(3) Within a few degrees of the position of equilibrium the righting moments should be comparatively small so as to give flexibility of control.

(4) As the displacement from the position of equilibrium increases the righting moments should increase also.

(5) The righting moments should never be excessive and should never exceed ⅓ of the possible moment which can be exercised by the elevator.

These results can be readily obtained by the use of a suitable negative tail. We shall now take a concrete case of an unstable orthogonal biplane with a total wing area of 432 square feet; wing section R. A. F. 6; the center of gravity of

TABLE 3.
TABLE ILLUSTRATING COMPUTATIONS IN STABILIZER DESIGN

Angle of wings to wind degrees	K_y corrected for biplane effect	K_x	Velocity of machine $=V$ (m.p.h.)	Resistance of wings in lbs. $D=K_x\ 432\ V^2$	Lift on machine in pounds $=L$	Arm for drag in feet $= b-b'=2$ ft.	Arm for lift in feet	Moment due to drag in lbs. ft.	Moment due to lift in lbs. ft.	Total moment for wings in lbs. ft.	Resultant angle of tail plane	K_n coefficient for flat plate aspect ratio 3	Arm for stabilizer in ft.	Lift on stabilizer in lbs.	Moment due to stabilizer in lbs. ft.	Resultant moment about c.g. in lbs. ft.
0	.000493	.000075	110.	392	2560	2	−1.2	+784	−3020	−2236	−3	−.00051	16	−309	+4950	2614
2	.000882	.000072	82.5	212	2560	2	− .6	+424	−1500	−1076	−2	−.00035	16	−119	+1910	834
4	.00121	.000093	70.	197	2560	2	− .30	+394	− 750	− 356	−1	−.00018	16	−44.1	+ 704	348
6	.00155	.000130	62.	216	2560	2	− .24	+432	− 600	− 168	0	0	16	0	0	− 168
8	.00187	.000166	56.6	230	2560	2	− .12	+460	− 300	+ 160	1	+.00018	16	+28.7	− 459	− 299
12	.00242	.000255	49.8	275	2560	2	0	+550	0	+ 550	3	+.00051	16	+65	−1040	− 490

Stabilizer set at −3 degrees to wing chord. Area of stabilizer = 50 sq. ft., Area of wings = 432 sq. ft.
+ moments tending to stall machine. − moments tending to dive machine.

the machine, as shown in Fig. 2. The loading is such that the machine has an angle of incidence of about 4 degrees at a speed of 70 miles per hour. The gap chord ratio is 1.0. The tail surfaces, stabilizer and elevator are taken together, are placed at a distance of 16 feet from the center of gravity. It is required to design a stabilizer to meet the above requirements.

We will assume that:

(a) The design of the stabilizing surface is carried out prior to wind tunnel tests.

(b) The center of pressure motion for each wing of the combination is precisely similar to that of the wing acting alone, so that L and D forces on each wing may be taken as acting at precisely the same point they would on a monoplane wing; this assumption is justified by results of both the biplane and triplane experiments.

(c) The resultant force on the tail surfaces may be taken as approximately perpendicular to the moment arm and as being equivalent to the lift. This is not far from true for angles within normal flight and simplifies our calculations enormously. The stabilizer surface is taken as being equivalent to a flat plate.

(d) The displacement of the vertical through the center of gravity for varying angles of incidence is neglected.

The machine selected for this problem is shown in Fig. 2 with the position of the center of gravity as indicated. It has a wing area of 432 square feet. The normal angle of flight being 4 degrees, it must weigh from elementary considerations (introducing .85 as correcting factor for K_y due to biplane effect).

$$W = K_y A V^2 = .0014 \times .85 \times 432 \times 70^2 = 2560 \text{ pounds}$$

Taking moments about the center of gravity the general equation for the pitching moment at any angle is

$$M = 2La - DC + DC^1 - 16R \qquad (1)$$

Where a is the distance between the vertical center of gravity line and the parallel line of lift on each wing (the wings assumed to carry equal loads), C and C^1 are the distances from the horizontal axis through the center of gravity to the point of application of the drag forces. The distance of 16 feet is assumed as the distance from the center of gravity to the point of application of the resultant force R acting on the stabilizer as defined previously.

Since, Fig. 2, the distance $C = 4$ feet; $C^1 = 2$ feet; the quantity $(b - b) = 2$ and the equation can be simplified to the following form:

$$2(La - D - 8R) = M \qquad (2)$$

The lift arm a varies of course with the center of pressure motion.

Unfortunately no set method of design exists for this particular problem. Area of stabilizer and the angle of setting to the wing chord have to be assumed more or less arbitrarily until the right combination is found, although each designer will probably find a short cut method. In two instances after a number of tentative combinations, the following values were found to specify our conditions fairly well.

Stabilizer, aspect ratio 3, area 50 square feet, angle of setting to wing chord 2 degrees. The K_y coefficients for the stabilizer treated as a flat plate can be taken from the curves given in Chapter 4.

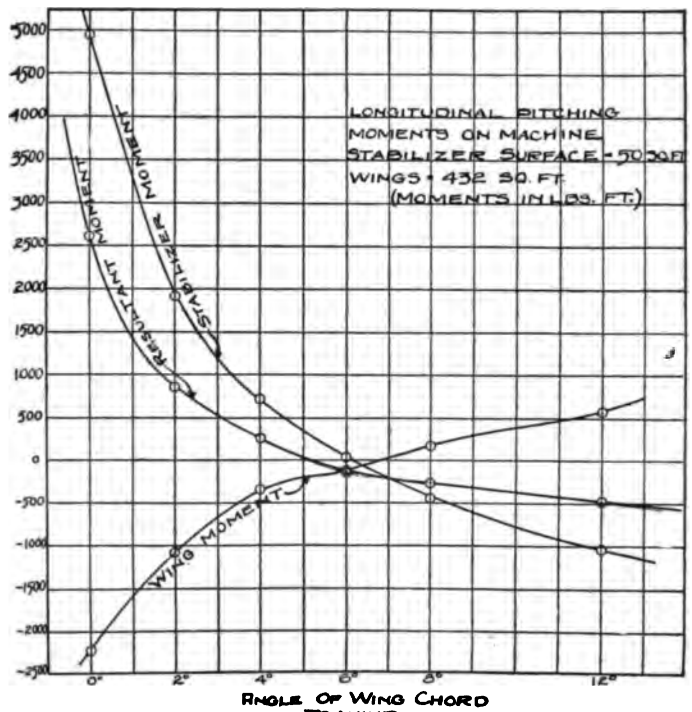


FIG. 3. LONGITUDINAL PITCHING MOMENTS ON MACHINE.

A careful distinction has to be made between the apparent angle of incidence of the stabilizer to the relative wind and the real angle, taking account of the deviation of the stream in accordance with the formula given. Thus if the wings are at an incidence of 6 degrees to the wind, and the stabilizer is at − 2 degrees to the wing chord, the real angle of incidence becomes

$$(6 - 2) - (\tfrac{1}{2}.6 + 1) = 0 \text{ degrees}$$

The computations for the setting finally selected are shown in the Table 3. The curves of moments due to the wings, the moments due to the stabilizer, and the resultant moment are shown in Fig. 3. The conventional method of regarding moments tending to stall the machine as positive, and moments tending to dive the machine as negative is adhered to.

It is seen from the curves of Fig. 3 that the solution is only fairly good. Equilibrium is secured at 6 degrees. Approximately on either side of the equilibrium position, the correct moments are obtained. If the machine goes below 6 degrees the resultant moment tends to stall it back to 6 degrees. If the machine goes above 6 degrees the resultant moment tends to dive it back to 6 degrees; that is satisfactory so far. But the resultant moment curve is far too steep about the position of equilibrium and the machine is sometimes too stable to be under perfect control. The problem is left in this form, however, because an improvement of the solution would be an excellent exercise for a student. The variations possible are in

I Length of stabilizer arm.
II Area of stabilizer.
III Angle of setting of stabilizer.

It must also be pointed out that the approximations employed, and the factors neglected, such as the effect of the body and other structural parts, are so numerous that in constructing an actual machine a wind tunnel test would be finally necessary.

Chapter X

Resistance of Various Airplane Parts

One of the most difficult problems in aeronautical design is the prediction of the total resistance of the machine. The wind tunnel test is a good check, but it is most important to assign resistance values to various parts and to tabulate them prior even to the construction of the model. In this chapter have been collected as far as possible all the data available for bodies, radiators, fittings, wheels, cables and wires and certain other miscellaneous objects.

Airplane Bodies from the Aerodynamical Point of View

If airplane bodies were designed from a purely aerodynamical point of view, they would follow dirigible practice and be of streamline form. There are, however, a number of structural requirements which have to be met, which preclude the employment of such forms. The body must enclose the power plant and the personnel, the length must be long enough to place the rudders well clear of the wash of the planes, the shape of the body must conform to structural requirements such as the use of four longitudinal girders, or a triangular form which has been found to be advantageous in steel construction.

No wind tunnel tests on bodies alone can determine exactly their resistance on an airplane, because the question is complicated by the position and form of the motor, and the disposition of the tail surfaces. The propeller in a tractor machine also introduces three possible variations in drag coefficients: (1) when the propeller is pulling and there is a slip stream of velocity greater than the airplane velocity, (2) the resistance on a glide when the engine is shut down, but the propeller is revolving as an air motor; (3) when the propeller is not revolving at all, the engine being held.

Tractor Bodies

In Table 1 is given a comparative table of resistance coefficients for area in normal presentation of a number of airplane bodies, and in Fig. 1 are shown sketches of the same bodies. Exact comparisons are impossible because some of the bodies are made for two men and others for one. Still qualitative conclusions can be drawn. The N. P. L. Model 5, more symmetrical than the B. E. 3, shows a distinct improvement over the latter which is somewhat discounted by the fact that the B. E. 3 carries two men unshielded. The B. F. 36, an almost perfect dirigible form, is markedly better than either of these two bodies.

The resistance of the body in an airplane is apparently a small quantity, but the figures given below do not represent the resistance of a body in full flight where it is increased by 40 per cent, the propeller slip stream increasing the relative speed of the air by some 25 per cent. Also, it must be remembered that with a best glide of 1 in 8, a 5-pound increase in resistance is practically equivalent to an added weight of 40 pounds. A blunt, square form of body such as is often seen in American practice may increase resistance even more,

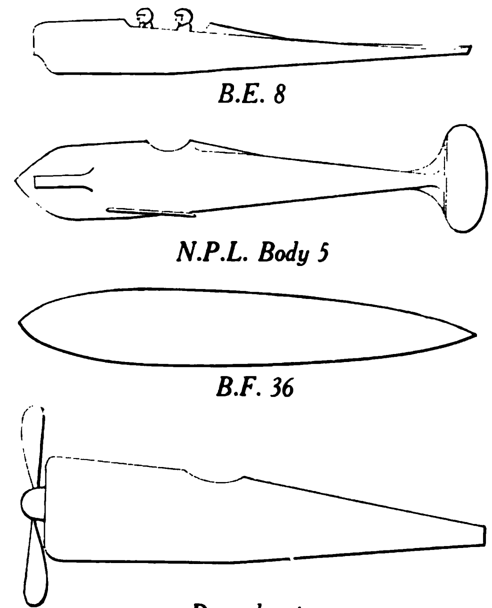


FIG. 1. TRACTOR BODIES.

and better aerodynamical design of bodies seems a feature worth considering.

TABLE 1.

COMPARATIVE TABLE FOR TRACTOR BODIES.

Designation.	Coefficient of resistance, K where $R = KAV^2$ (A = area in normal presentation in square feet V = miles per hour; R = drag in lbs.)	Length / maximum depth.	Resistance for a body of 8 square feet normal presentation at a speed of 60 m. p. h.
British B. E. 3 (with 2 men)	.000720	7.35	20.7
N. P. L. Model 5	.000420(approx.)	5.50	12.0
British B. F. 36 (dirigible form)	.000258	5.75	7.4
Deperdussin (enclosing rotary motor)	.001215	5.6	35.1

TABLE 2.

COMPARATIVE TABLE FOR PUSHER BODIES.

Designation.	Coefficient of resistance K where $R = KAV^2$ (A = maximum area in normal presentation in square feet; V = miles per hour; R = drag in lbs.)	Length.	Resistance for a body of 8 square feet normal presentation, at a speed of 60 m. p. h.
N. P. L. Model Body 3 (fairly symmetrical section)	.000271	3	7.8
Farman 3 (body in form of a boat, two men unshielded)	.000845	3.2	24.4

Pusher Bodies

A pusher body such as the Farman 3, illustrated in Fig. 2, gives a not much larger resistance than the tractor bodies, but when the head resistance of the uncovered outriggers is taken into account, it will probably be found that

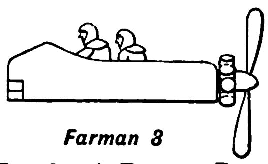


FIG. 2. A PUSHER BODY.

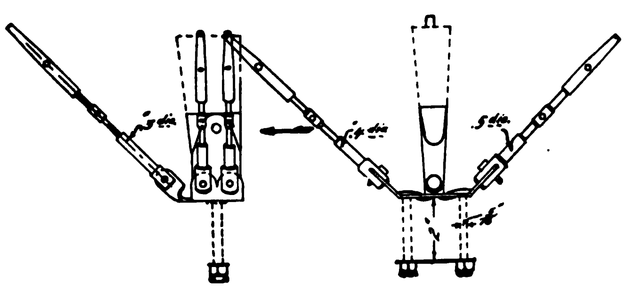
pusher arrangements offer considerably more resistance than tractor bodies.

Radiator Resistance

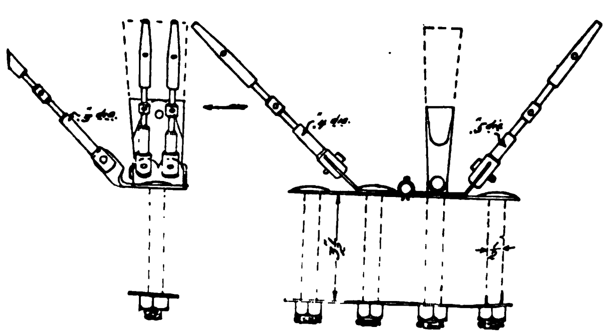
The only values available for this are the results of some tests at the Massachusetts Institute of Technology. These were carried out on portions of a radiator of the honeycomb type having sixteen ¼-inch cells to each square inch of the surface normal to the wind. The tests were repeated on two sizes of radiator section, one 0.25 square feet and the other 0.111 square feet, and at various wind speeds. No important variation in the resistance coefficient was apparent and the average coefficient may be used for practical calculations. This has a value $K_x = .000814$ pounds per square foot of projected area per foot per second or .00173 pounds per square foot of projected area per mile per hour.

Resistance of Fittings

Fittings are so variable in design that it is impossible to give definite figures to meet every type of wing strut fitting. Tests were conducted at the Massachusetts Institute of Technology on the fittings of which dimension drawings are given in Fig. 3; the coefficients of resistance are $R = .00030\ V^2$ and $R = .00040\ V^2$ for the two types which at 60 miles an hour



Outer Panel. $R = .00030\ V^2$.



Inner Panel with Wing Hinge. $R = .00040\ V^2$.

FIG. 3. FITTINGS EMPLOYED IN TESTS FOR HEAD RESISTANCE AT MASSACHUSETTS INSTITUTE OF TECHNOLOGY

CLARK STRUT FITTINGS. Resistance includes fitting, five turnbuckles and nuts but not dotted portions as indicated on drawings. Resistance in pounds; Velocity in miles per hour.

gives 1.07 and 1.44 pounds respectively. Such figures will be at least approximately correct in design.

Resistance of Airplane Wheels

For a standard airplane wheel of about 26 × 4 inches in size, the drag found by the N. P. L. is about 1.7 pounds at 60 miles per hour. This again is sufficiently accurate for practical purposes. Eiffel has experimented with a number of wheels and shown that no great variation need be expected from the above value. An important result from the French experiments was the fact that an uncovered wheel had a resistance of 50 per cent more than a covered wheel of similar dimensions. This justifies the standard practice of covering the wheel in.

Resistance of Wires and Methods of Plotting

A certain complication is necessary in the methods of plotting the results for the resistance of cables and wires. As we have seen from the diagram of Fig. 19, in Chapter 3, of the Course, the resistance of a wire or any cylindrical body is partly due to turbulence, partly due to skin friction. It cannot therefore be represented by such a simple expression as $R = KAV^2$ as for a wing, but by an expression involving Reynolds' number, $R = KAV^2 f\left(\frac{\nu}{DV}\right)$, or if we replace the area of a wire by LD where D = diameter and L is length in feet, then $R = KLD\ V^2 f\left(\frac{\nu}{DV}\right)$. Since ν, the coefficient of kinematic viscosity is constant for air, we can simplify this expression by writing:

$$R = KLD\ V^2\ F(VD).$$

We do not know what the function $F(VD)$ is exactly, nor how it varies with size and scale except from experimental results, and comparisons of resistance varying as $LD\ V^2$ can only be made between two cables if VD is a constant. If K is taken as a function of VD, then R may be written $R = KLD\ V^2$ but then K must be plotted against VD in analyzing experimental results. This is the only rational and scientific method.

An empirical method, however, is sometimes employed with fair accuracy of plotting the resistance of a wire whose length is equal to its diameter against $V^2 D^2$. This has the advantage that the graph approximates very closely to a straight line, the slope of which is equal to K, thus giving an easy means of determining a mean value of K.

Resistance of Stationary Smooth Wires

The most accurate researches have been carried out at the N. P. L. and their results are shown in Fig. 4 plotted against $V'D$. In the expression $K = \frac{R}{LDV^2}$, R is in pounds, L in feet, D in feet, and V in miles per hour. But in the abscissae, values of $V'D$, V' is in feet per second, and D in feet, so as to give the correct scale and speed relationships which must be in the same units.

The accuracy of the curve at its lowest portion is doubtful, since the flow is apparently just changing its nature at that point, and successive observations under the same conditions may give quite different results. On modern machines of fairly high speed, however, the values of $V'D$ nearly always exceed 0.35 and consequently, do not lie on this section of the curve.

Similar tests were made by Mr. Thurston and M. Eiffel, and the values obtained by the former are plotted in the same figure. Thurston's experiments, however, were very much earlier, and Eiffel's covered a less range and were performed

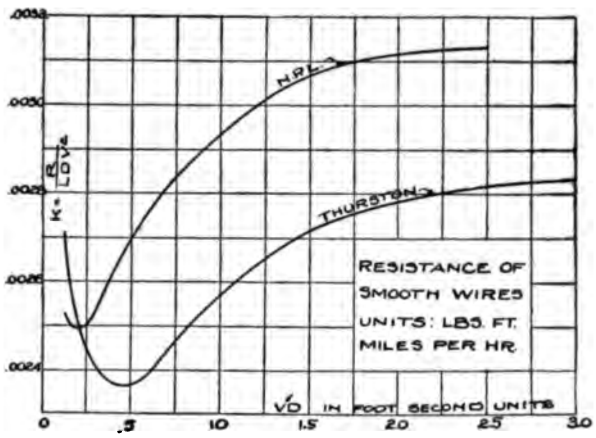


FIG. 4. RESISTANCE OF SMOOTH WIRES PER FOOT RUN.

with less sensitive apparatus, so it is advisable to use the N.P.L. results.

Resistance of Vibrating Wires

When the question of resistance first began to arouse interest, it was popularly supposed that a vibrating wire had much greater resistance than a stationary one. This, however, is not the case. Research on this point at the N.P.L. failed to disclose any difference whatever, although the balance would have shown deviations as small as 3 per cent, even for the extremely small forces under consideration. Mr. Thurston, on the other hand, concluded that vibration at the rate of 15 per second increased the resistance by about 5 per cent for small wires and by a somewhat smaller percentage for those of larger diameter. In any case, the effect is unimportant.

Resistance of Stranded Wires

The air resistance of stranded wires was also investigated at the N.P.L., and was found to be about 20 per cent greater than that for a smooth wire of the some diameter. This is only approximate, as the coefficient depends on the number of strands, type of lay, etc. It is also impossible to plot the values of K against VD for wire rope, as the VD law holds good only for objects which possess strict geometrical similarity, a thing which stranded wires of different sizes never do.

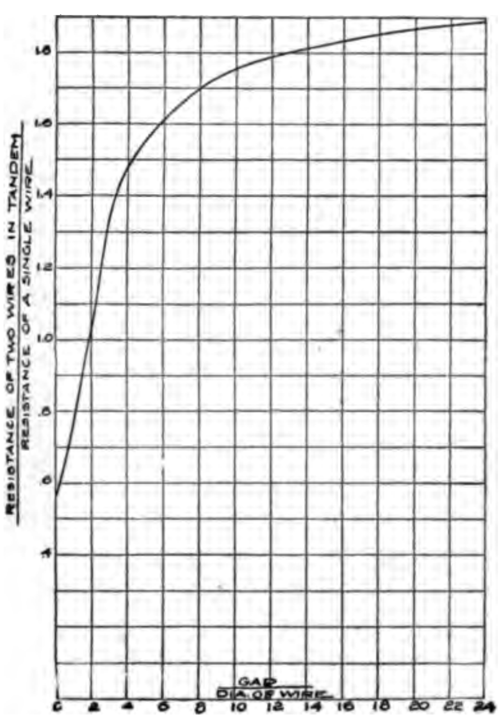


FIG. 5. RESISTANCE OF WIRES IN TANDEM AS A RATIO OF THE RESISTANCE OF A SINGLE WIRE.

Resistance of Wires Placed Behind One Another

The manner in which resistance is affected by the close juxtaposition of two wires, one behind the other, is a point of great interest. Here, too, it is at present necessary to rely on Mr. Thurston, although we hope to be able soon to present the results of some more extensive and accurate tests on this matter.

Fig. 5 gives, in terms of the resistance of a single bar, the resistance of two bars or wires separated by various distances. It will be seen that two wires placed one behind the other and spaced from 5 to 9 diameters apart, as is usual in double-wiring a biplane cellule, have from 60 per cent to 75 per cent more resistance than a single wire. The force is,

OBJECT	K for pounds per square foot per mile hour units	LIMITS VD in foot second units	ATTITUDE
Sphere	0.000445	$VD > 32$	
Hemispherical Shell	0.003840	$VD > 11$	
Hemispherical Shell	0.008100	$VD > 22$	
Circular Disk	0.002820	$VD > 22$	
Cone Closed Base	0.001300		
Cone Closed Base	0.000850		
Cone Hemispherical End	0.000406		
Cone Hemispherical End	0.000222		

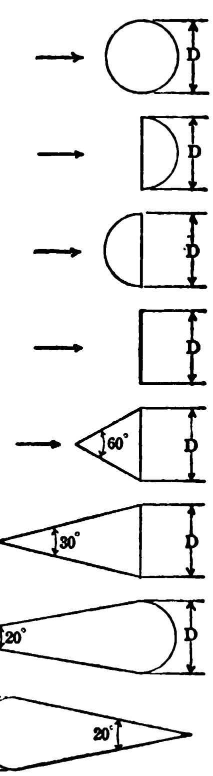


FIG. 6. RESISTANCE OF MISCELLANEOUS OBJECTS (AFTER EIFFEL)

however, materially less than for the two wires placed side by side.

Resistance of Inclined Wires

Eiffel has experimented on the resistance of inclined wires. As would be expected the resistance of a wire progressively decreases as its angle with the wind diminishes. Table 3 gives correcting values.

TABLE 3.

Angle of a wire to the wind.	Ratio of resistance to that of a wire at 90 degrees to the wind.
90 degrees.	1.00
75 degrees.	0.92
60 degrees.	0.70
45 degrees.	0.46
30 degrees.	0.20

Suggestions for Stream-lining Wires

It has been suggested from time to time that wire resistance should be decreased by "stream-lining" or adding a triangular portion in back of the wire. From experiments by Ogilvie, however, it appears that a section made up of a semi-circle and a triangle has a decidedly high resistance, and the gain from such a procedure would be small.

Wires placed behind one another have also been covered in. The British Royal Aircraft Factory produces a very heavy R.A.F. wire in use on big machines which is stream-line in form. But the direction in which progress manifests itself at present is in the elimination of wires by certain modern trussing such as used in the recent Curtiss biplane.

Resistance of Miscellaneous Objects

The resistance of certain miscellaneous objects as deduced by Eiffel may sometimes be useful. The values for such objects within certain limits are illustrated in Fig. 6.

References for Part I, Chapter 10

AIRPLANE BODIES

British Report 1911-1912, page 52.
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"La Resistance de l'Air et l'Aviation," Eiffel, 1914, page 250.

AIRPLANE WHEELS

British Report 1912-1913, page 122.
"La Resistance de l'Air et l'Aviation," Eiffel, page 250.

WIRE AND CABLES

"Aerodynamic Resistance of Struts, Bars and Wires," by A. P. Thurston, *Aeronautical Journal,* April and July, 1912.

British Report 1910-1911.

"La Resistance de l'Air et l'Aviation," Eiffel, page 97.
"New Mechanical Engineers' Handbook," Section on Aeronautics, by J. C. Hunsaker.

Chapter XI

Resistance and Comparative Merits of Airplane Struts

Considerations of Comparative Merit of Strut Sections

It is naturally desirable that some single expression be found which will give the general efficiency and theoretical desirability of any strut section under consideration. This was first done by the staff of the National Physical Laboratory, who devised what they called the "equivalent weight," and we have here employed an adaptation of this quantity under the name of the "merit factor." In deriving this, we start with the basic assumptions that the speed of the machine is 60 miles per hour, the gliding angle 1 in 7, and the average width of the struts about 1 inch (the exact breadth assumed depending on the form and strength of the section).

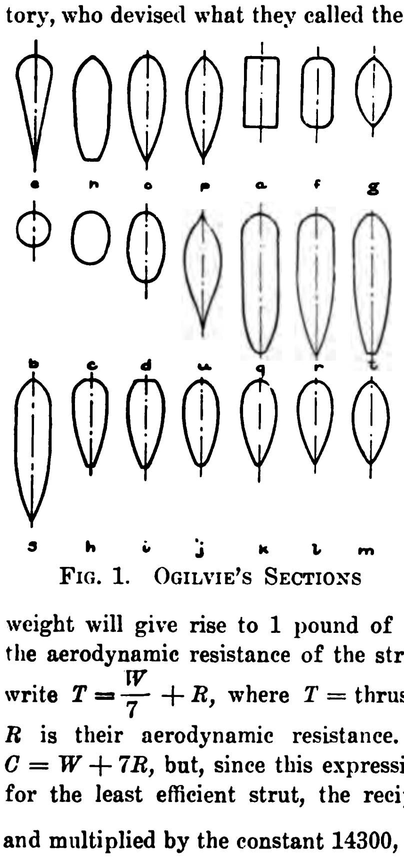

FIG. 1. OGILVIE'S SECTIONS

Then, since gliding angle $= \frac{W}{R}$, every 7 pounds of strut weight will give rise to 1 pound of resistance, in addition to the aerodynamic resistance of the struts.† We can, therefore, write $T = \frac{W}{7} + R$, where T = thrust due to the struts, and R is their aerodynamic resistance. Simplifying, we have $C = W + 7R$, but, since this expression has a maximum value for the least efficient strut, the reciprocal is here employed, and multiplied by the constant 14300, giving $C = \frac{14300}{W + 7R}$

The best strut under the conditions above specified is then the one showing the highest value for C. The reason for choosing this particular value for the multiplier is that it makes $C = 100$ for the best strut of the first and largest series which we shall consider.

If the speed of the machine for which the struts are being selected is greater than 60 miles an hour, the resistance becomes of greater importance as compared with the weight, and the merit factors for those sections which, although heavy, offer very low resistances are relatively improved. If the gliding angle is flatter than 1 in 7, a similar effect ensues.

On the other hand, if it becomes necessary to use struts having a diameter of more than 1 inch or thereabouts, the advantage inclines toward the sections which have the greatest strength for their weight, and the relative importance of resistance is diminished, since, in similar sections, weight varies as the square of the breadth and resistance only as the first power. These effects are, however, of slight importance, and would not be likely to change the merit factors enough to have serious influence on the choice of a section in any given case.

The question of strength will be taken up more fully in another section of the course. It will suffice to say here that the strengths of two struts have been considered to be equal when their moments of inertia about their longitudinal axes are equal.

Strut Sections Developed by Ogilvie

We may now proceed to the examination of definite data for a number of series of struts, tested at various times and places. The following figures are the result of experiments performed at the N. P. L. at the suggestion of Alec Ogilvie, the sections being illustrated in Fig. 1.

I = moment of inertia for the section in question about its longitudinal axis (inches[4] for a strut 1 inch wide).

R = resistance in pounds of 100 feet of strut 1 inch wide at 60 miles per hour.

W = weight in pounds of 100 feet of spruce strut 1 inch wide.

b = width of strut whose strength will be equal to that of a strut of section a, and 1 inch wide.

W' = weight of 100 feet of spruce strut of width b.

C_{60} = merit factor at 60 miles per hour.

No.	I	R	W	b	W'	C_{60}
a	.167	104.4	41.6	1.00	41.6	19
b	.049	81.9	16.4	1.36	30.3	18
c	.090	59.2	30.4	1.17	41.6	27
d	.124	36.9	34.8	1.08	40.6	45
e	.074	63.0	33.4	1.23	50.6	24
f	.134	28.6	37.7	1.06	42.4	56
g	.094	54.9	30.0	1.15	39.7	30
h	.119	12.8	39.7	1.09	47.1	99
i	.127	12.8	41.0	1.07	47.0	100
j	.119	13.5	39.7	1.09	47.1	96
k	.111	13.5	38.0	1.11	46.8	94
l	.106	29.9	36.4	1.12	45.6	51
m	.106	45.9	36.6	1.12	45.9	35
n	.171	14.2	51.9	0.99	50.9	97
o	.146	13.5	47.0	1.03	49.9	97
p	.128	18.7	44.1	1.07	50.5	75
q	.245	15.1	71.0	0.91	58.9	93
r	.227	16.4	67.2	0.93	58.1	87
s	.194	13.5	62.0	0.96	57.2	97
t	.209	13.5	66.1	0.95	59.7	95
u	.115	24.6	42.5	1.10	51.4	59

Many very interesting conclusions can be drawn from this table. In the first place, it is evidently of the utmost importance to avoid rapid changes in curvature. Several sections, notably, *e* and *l*, although they appear to have a very smooth outline, oppose a large resistance simply because the transition from the entrance to the run is so abrupt that the air-flow cannot follow its contour, and violent eddy-making ensues.

† Relationships between weight and resistance on a glide will be fully considered in Section 12.

The good performance of several sections so formed indicates that it may be wise actually to run the sides of the strut parallel for some little distance, as illustrated by *q* and *t*. This is counteracted, however, by the fact that skin-friction increases in proportion to the "wetted surface" of the strut. It is for this reason that the very longest sections did not give such low resistances as those of more moderate form. This matter of the ratio of length of section to width will be discussed more fully somewhat later, in connection with another series of tests.

It will be seen, too, that the resistance is little affected by the chopping off of a portion of the tail in such a manner as to leave it straight across. Examples of this are furnished by *n*, *t* and *i*. This is due to the fact that it has not been possible in any strut yet designed to totally eliminate the region of deadwater behind the strut. As will be evident from any section of air-flow about a fair-shaped section, the lines of flow always leave the contour of the strut some distance short of the extreme rear. Since no changes made in the contour within this region will have any decided effect on the resistance, it avails nothing to go to the trouble and expense involved in the attempt to construct a wooden strut running out to a sharp point at the back.

Another Series of Struts Tested at the N. P. L.

At about this same time another series of struts was tested at the same laboratory, the sections being those actually em-

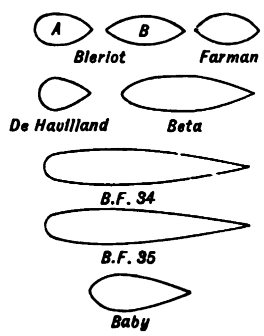

Fig. 2. Strut Sections Tested at N. P. L.

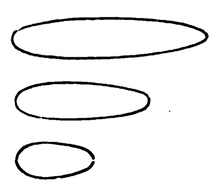

Fig. 3. N. P. L. Struts

ployed in machines then existing. The outlines of the sections tested are shown in Fig. 2, and the characteristics are given below.

Name.	*I*	*R*	*W*	*b*	*W'*	C_{60}
Bleriot A....	.070	51.0	26.0	1.24	40.0	30
Bleriot B....	.107	52.7	34.9	1.12	43.8	31
Farman	.074	49.3	25.2	1.22	37.5	31
De Havilland.	.052	54.9	20.5	1.34	36.8	26
Baby	.110	17.0	41.6	1.11	51.5	78
B.F. 34......	.279	15.5	93.2	0.88	72.0	85
B.F. 35......	.238	13.5	89.7	0.92	76.0	88
Beta	.188	14.8	61.7	0.97	58.0	90

It will be seen that these figures simply supplement and confirm the conclusions already deduced from the more extensive and systematic investigations directed by Mr. Ogilvie.

Tests on Struts, Length to Width Varied

As a result of these and other tests, a series of struts embodying the best features of those already tried, and varying only in the ratio of length of section to width, was made and tested at the National Physical Laboratory. Three representative members of this series are shown in Fig. 3. The table below gives the characteristics of these struts, the meaning of the symbols being the same as in the tables already given, except that n = the ratio of the length to width of section.

n	*I*	*R*	*W*	*b*	*W'*	C_{60}
2.	.094	24.8	32.0	1.15	42.3	59
2.5	.117	13.7	40.0	1.09	47.5	94
3.	.141	13.4	48.1	1.04	52.1	96
3.5	.164	11.4	56.1	1.00	56.1	105
4.	.188	11.2	64.1	0.97	62.2	103
4.5	.211	11.7	72.1	0.94	67.8	99
5.	.235	12.1	80.1	0.92	73.7	94

Thus it is apparent that the best of these sections are materially superior to the best of the sections tested by Ogilvie, both in resistance and in merit factor. In Fig. 4 resistance of

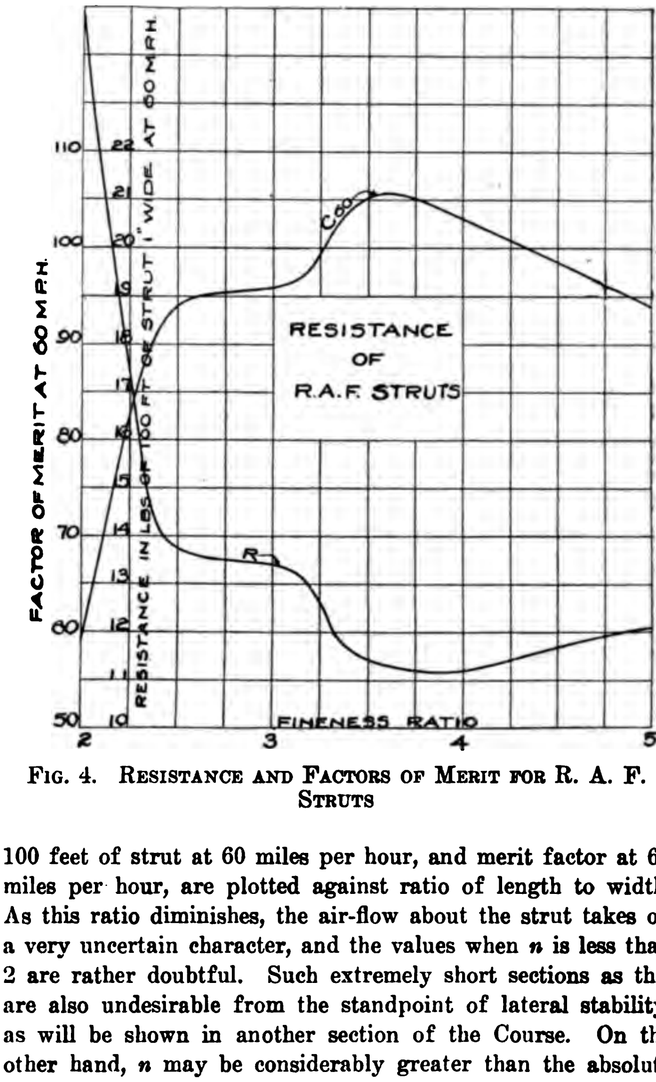

Fig. 4. Resistance and Factors of Merit for R. A. F. Struts

100 feet of strut at 60 miles per hour, and merit factor at 60 miles per hour, are plotted against ratio of length to width. As this ratio diminishes, the air-flow about the strut takes on a very uncertain character, and the values when n is less than 2 are rather doubtful. Such extremely short sections as this are also undesirable from the standpoint of lateral stability, as will be shown in another section of the Course. On the other hand, n may be considerably greater than the absolute optimum value without any great disadvantage, so it will be well in general to employ a ratio of four, or even a slightly higher figure. The photographs of flow about strut sections, reproduced in Fig. 5, show clearly why such a procedure can be safely adopted.

Two Eiffel Struts

Two struts of somewhat the same section as those just discussed have recently been tested by Eiffel, and show remarkably low resistances. Their outlines are given in Fig. 6. For No. 1, having n equal to 3.23, R equals 9.7 pounds, while for No. 2, with a somewhat sharper entry, n is 2.96 and R is only 8.7 pounds. Part of this improvement over the best of the English tests, however, is undoubtedly due to the higher wind speed which is secured in Eiffel's laboratory, the resistance coefficient having a tendency to rise as the speed of test is decreased.

Effect of Length of Struts

We now turn our attention to the effect of the length of the strut. While this point is less important than was generally supposed a few years ago, and while its effects are largely determined by the nature of the surfaces in which the strut terminates, the experimental results bearing on the matter should nevertheless be studied. For this data we are indebted to Mr. Thurston, who has described his results in the series of articles already cited. As the result of a great

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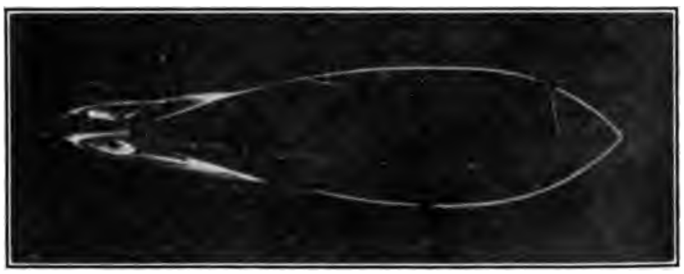

BETA

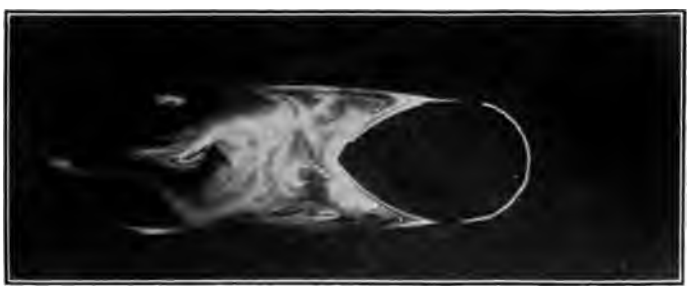

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FIG. 5. ILLUSTRATING FLOW AROUND STRUTS

many experiments on manifold different types of strut, he came to the conclusion that resistance for a strut with free ends could best be expressed by the formula $R = KltV^2 - .0073t^2V^2$, where R is the resistance in pounds, l and t, respectively, the length and thickness of the strut in feet, K a constant, and V the speed in miles per hour.

It is evident from this equation that, even with the lowest values of K yet obtained, the effects of length will be prac-

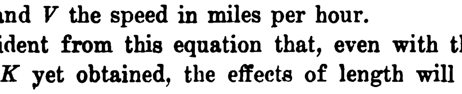

FIG. 6. TWO EIFFEL STRUTS

tically negligible when the length is more than 50 times the thickness, as it generally is. Since, in addition, the case of a strut with free ends is one which never occurs in practise, resistance may be considered as independent of length-thickness ratio for all the purposes of design.

The form of air-flow about the wing may have very decided effects on the resistance of interplane struts, but we have no means of knowing how great these are, and experiments covering this point and susceptible of performance in a wind tunnel would be exceedingly difficult to devise. The matter might well be investigated in an outdoor, full-scale plant such as that at St. Cyr.

Resistance of Inclined Struts

The only point which remains to be studied is the resistance of struts which are not normal to the line of flight. Some much more recent tests by Mr. Thurston have covered this point, and show very surprising results. Struts of square, rectangular, circular, and stream-line section were tested at angles from 0 to 90 degrees, and the effects of the ends of the strut offering a direct resistance when inclined were overcome by the use of the method of differences: that is, tests were made first on a strut 34 inches long, and then on one 16 inches long, the difference of the figures obtained being equal to the resistance of an 18-inch section of an infinite strut.

The ratio of the resistance of a strut inclined at various

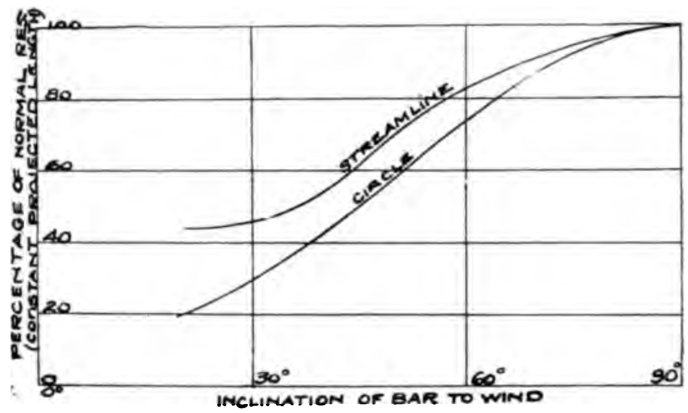

FIG. 7. DATA FOR INCLINED STRUTS

angles to the resistance of a normal strut of like section and equal projected length is plotted in Fig. 7. It will be seen that the resistance at 30 degrees to the wind is less than one-third of that at 90 degrees, and this large difference is by no means accounted for by the difference in length of section parallel to the wind. When a circular strut is placed at an angle of 30 degrees to the wind, the section parallel thereto is an ellipse having a length of twice its width, and the resistance of an elliptical strut such as this, when placed normal, is only 36 per cent less than that for a circular section. About 45 per cent of the reduction due to inclination thus remains unaccounted for.

Since, however, the curve of reduction is substantially a sine curve, and is therefore very flat at the ends, there is very little advantage to be gained from inclining a stream-line strut unless it is inclined at least 30 degrees to the normal. This reduced resistance should, however, be kept in mind as a point in favor of the staggered biplane. Eiffel also made a few tests on struts inclined 30 degrees from the normal, the results checking very well with Mr. Thurston's.

The Effect of Changing the *DV* Product for Struts

As was shown in Chapter 10, the resistance coefficient is not an absolute constant, but is a function of VD, where V is the speed and D the diameter of the strut. The coefficient tends to decrease as VD increases, but the change for values of VD (in foot/second units) above 6 is extremely small, as Eiffel has demonstrated. The tests made at the National Physical Laboratory have been made with a value of VD equal to only 2.5, whereas, in an actual machine, this quantity would never be likely to fall below 5, and is generally from 7 to 10.

We can therefore deduce from Eiffel's experiments that it is safe to reduce the values for resistance here given (for the N. P. L. tests) by about 25 per cent in applying them to a design. This indicates that, as was hinted above, the superiority of Eiffel's strut sections is more apparent than real, and that the best sections yet available are the N. P. L. sections having fineness ratios of from 3.5 to 4.5. The correction given here should be applied only to struts of fairly good section, as the value of VD has much less effect on those sections for which the resistance is relatively high, and in which there is more effect due to turbulence than to skin friction.

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Chapter XII
Resistance and Performance

Nomenclature

It may be useful to restate the symbols which we employ in considering performance curves, ascent and descent.

W = weight of the machine.
A = area of the wings.
i = angle of incidence of the wings.
L = lift.
K_y = lift coefficient.
D = drag of wings.
K_x = drag coefficient.
R = resultant of lift and drag on the wings.
P = parasite or structural resistance of a machine.
D_t = total resistance or drag = $D + P$.
R_t = total resultant air force on a machine.
H = propeller thrust.
θ = angle of flight path with the horizontal.

Structural and Wing Resistance for the British B.E.2

In Chapter 4, a problem was worked out on the sustentation and resistance of wing surfaces, which in spite of some rough

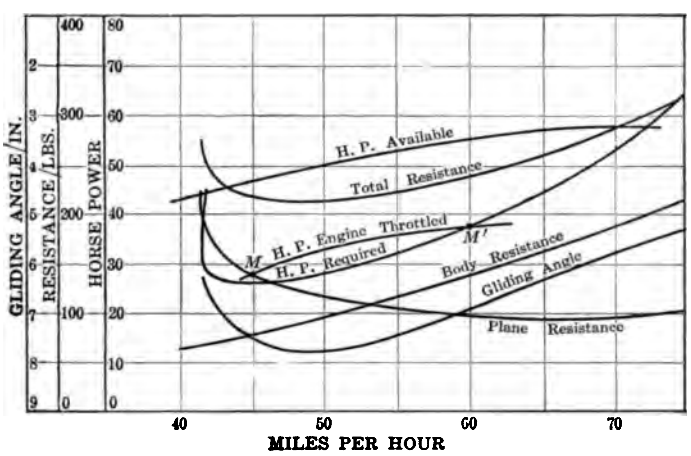

FIG. 1. PERFORMANCE CURVES FOR THE B.E.2

assumptions, illustrated the main performance curves and calculations employed. In Fig. 1 are shown curves for the British B. E. 2. It is not a particularly modern machine, but has been worked out so thoroughly that it deserves particularly careful study.

The body or parasite resistance which includes the resistance of the wing bracing, chassis, etc., as well as the resistance of the body proper, is taken as varying as V^2 and allowance has been made for propeller slip stream velocity. The body resistance is seen to play an unimportant part at low speeds. But at about 53 miles per hour it becomes greater than the plane or wing resistance, and at high speeds it is almost twice as great as the wing resistance. This emphasizes the importance of minimizing the resistance for a high-speed machine. However good a wing section itself may be, high structural resistance will make high speeds impossible.

The plane resistance curve has a minimum value at about 65 miles per hour and increases on either side of this speed. It is interesting to follow out how this increase in resistance on either side occurs. At high speeds, the angles of incidence and the drift coefficients are small but the speeds are very great, and the increase in wing resistance is obvious. At small speeds on the other hand the airplane is flying at large angles of incidence to give the necessary sustentation and the drift coefficients are large. The shape of the total resistance curve follows from the summation of the two.

Theoretical Laws for Minimum Thrust and Minimum Horsepower

From a theoretical treatment of the question, the following interesting law has been derived:

Minimum thrust is required to overcome the resistance of an airplane when the parasite resistance is equal to the drag of the wings.

For a proof of this law, reference to Chassériaud and Espitallier is appended. In the case we have selected, illustrated in Fig. 1, the structural air resistance and the wing drag are equal at a speed of 53 miles an hour, while the minimum resistance is at 49 miles per hour. The law does not seem to be borne out by practice, though it may be occasionally useful as a rough check.

The minimum horsepower required generally occurs at a low speed, but not at the minimum speed; and its position will vary for every machine. Another theoretically deduced law states that:

Minimum horsepower is required when the machine is moving at a speed at which the wing resistance is three times the body resistance.

This law is often highly inaccurate, but may be useful.

Effective or Propeller Horsepower Available Curve

Typical curves for these are also illustrated in Fig. 1, and are of the greatest interest to the designer. In establishing such curves it is generally assumed that the engine is running at the rated revolutions per minute and that in designing the propeller the efficiency for this revolution per minute at every airplane speed is known. Thus assuming an engine which delivers 140 horsepower at an airplane speed of 80 miles an hour, the propeller having an efficiency of 75 per cent at this speed, the available horsepower will be

$$\frac{140 \times 75}{100} = 105 \text{ horsepower.}$$

Since the power of a propeller is given by the product of its thrust into the speed and the speed of the propeller is the speed of the airplane, it follows that when the propeller is delivering sufficient power, it is also delivering sufficient thrust. Hence propeller horsepower available is sufficient for all practical consideration, and propeller thrust curves need not be included in a performance chart.

Minimum and Maximum Speed; Maximum Excess Power; Best Climb; Descent

The maximum and minimum speeds of an airplane are generally given by the two points of intersection of the propeller horsepower available and the total horsepower required. If the machine is highly-powered, and the propeller efficient, the two curves may not intersect at the speed at which the lift becomes insufficient, and the airplane would climb at stalling angle, unless the engine is considerably throttled down. The climb decreases the angle of incidence, and checks stalling. It is thus a decided advantage to have excess available power at high angles.

It is a simple matter to deduce the speed of climb from the excess power. This is absorbed in raising the machine.

$$\text{Excess power} = \frac{\text{Total weight} \times \text{climb per second}}{550}$$

The maximum excess power does not occur at the lowest speed. To find it, we must measure the maximum ordinate between the available propeller horsepower and the total required horsepower. In Fig. 1 this is to be found at 48 miles per hour. The excess is 21 horsepower and the weight of the machine is 1650 pounds.

$$\text{Climb} = \frac{21 \times 550}{1650} = 7 \text{ feet per second or 420 feet per minute.}$$

This is, however, only the initial rate of climb. As the machine rises, the density of the air, the power of the engine, and the climb gradually diminish.

In practice, the pilot need not know the change of incidence that he produces to climb, although for a given machine it is an easy matter to calculate the correct angle from the performance curves. In Dr. Hunsaker's words, "a careful man moves his elevator slowly until he has placed himself on the desired trajectory." Part of the art of aviation is to do this without exceeding safe limits, for obviously there is a limit to the rate of climb the engine can handle. If the machine is put on a climb too steep for the power of the machine, the speed is suddenly lost, the controls become ineffective, and the machine has stalled.

In descent, very analogous considerations obtain. The pilot decreases his angle of incidence to a negative value. At this angle the speed required for sustentation is beyond that of the maximum, and the propeller horsepower is insufficient. If D = deficiency in horsepower,

$$D = \frac{\text{Total weight} \times \text{velocity of descent.}}{550}$$

The machine descends and gains the required speed under the action of gravity.

The Two Regions of Control. Control by Throttling

Consider the performance curves of the same machine, the British B.E.2 shown in Fig. 1. Suppose the machine to be flown at 10 degrees at the point *M* with the engine throttled, so that there is equilibrium, and the power curve is as shown, 26 horsepower. The pilot wishing to rise will naturally increase his angle of incidence to say 12 degrees. He will then require 30 horsepower while the throttled engine will deliver even less than the 26 horsepower through the propeller. Instead of rising the machine will fall.

Suppose now that flying at the same point and under the same conditions he wishes to descend, and decreases his angle to 8 degrees. He will now have an excess of power of 3 horsepower as can be seen from the curves and will ascend instead of descend. There is therefore a region of reverse controls, known to French authors as the *régime lent.*

At the point M^1, when the pilot wishes to rise and increases his angle of incidence, he does indeed obtain excess power and rises. Here the controls are normal and the region is known as *régime rapide.* For an inexperienced pilot the *régime lent* is dangerous. Even if he knows the angle of incidence at which he is working, he is likely to get into difficulties.

With a flexible engine, an expert pilot can operate an airplane in the slow speed region by manipulation of the throttle

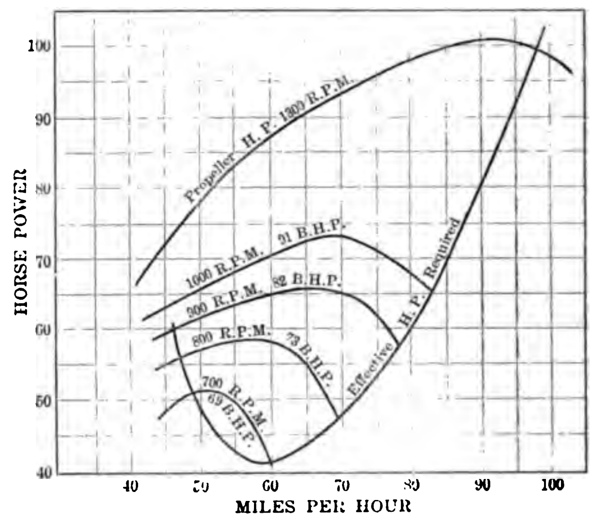

FIG. 2. VARYING SPEED RANGE WITH ENGINE THROTTLED

alone. In Fig. 2 the propeller horsepower available is shown with the engine throttled down to various speeds for a design taken from Dr. Hunsaker's pamphlet, to which reference is appended. For each speed of the engine there is a different maximum and minimum speed of the airplane, and a different speed range. If the airplane is flying at the minimum speed in the *régime lent* region at a certain revolution per minute, the pilot can by unthrottling his engine pass to a larger speed range, obtain excess power and climb without changing his forward speed or angle of incidence. When an engine is throttled the danger of reversed controls is still greater, because the speed range becomes so very small. Even the best of pilots may mistake his position on the curve.

In French airplane contests, a premium has been placed on low speeds, and the *régime lent* with throttling has been largely and successfully used. Such operation does not seem advisable for ordinary flying.

Variations in Propeller Horsepower Curves

We will now consider the possible variations in performance by changing the design of the propeller from a high speed to a climbing propeller. In Fig. 3 the B.E.2 is again illustrated. The power required curve remains the same. By suitable design the propeller efficiency curve can be changed so as to give maximum efficiency at varying speeds. The design of a suitable propeller cannot unfortunately be detailed here.

For the propeller with Efficiency Curve 1, the maximum efficiency is at high speed, and the Horsepower Available Curve 1 shows that such a propeller will give a high maximum speed. It is a high speed propeller when applied to this particular airplane.

For the propeller with Efficiency Curve 2, the maximum efficiency occurs at a lower speed. Such a propeller will give

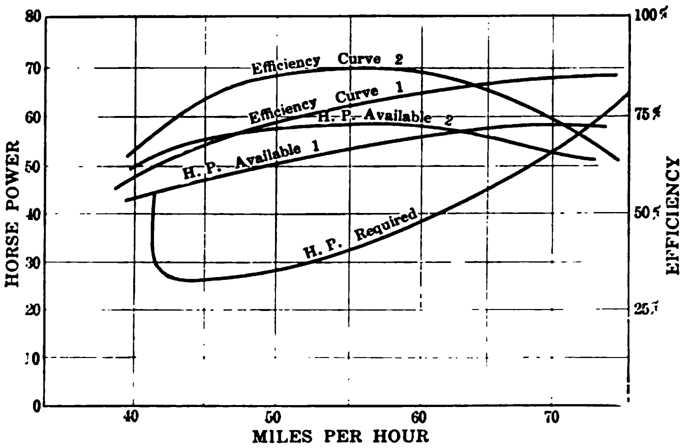

FIG. 3. VARIATION OF PERFORMANCE WITH CHANGE IN PROPELLER DESIGN

a smaller maximum speed, as can be seen from Horsepower Available Curve 2, but a greater excess power. It will be a climbing propeller. There are many such variations possible for any machine.

Angle of Glide

The best L/D for a wing section may be in the neighborhood of 14 or 15. But the parasite resistance of a machine, *i. e.*, the resistance of the body, wing bracing, etc., increases the drag to such an extent that the L/D_t of the whole machine may be reduced to 7 or 8. It is this value of L/D_t which determines the angle of glide of a machine.

In Fig. 4 is shown a machine which is gliding with the engine shut down so that the propeller exerts no thrust, i being the angle of incidence, and θ, the angle which the machine makes with the horizontal line, being the angle of glide. Resolving forces perpendicular to and along the line of motion, the equations of equilibrium for steady glide are:

$$W \cos \theta = L \quad (1)$$

$$W \sin \theta = D_t = D + P \quad (2)$$

The angle of glide is therefore given by the equation

$$\tan \theta = \frac{D_t}{L} \quad (3)$$

and has its maximum value when D_t is a maximum.

The minimum angle of glide is also termed the "best" angle of glide. At a given height above the ground, the

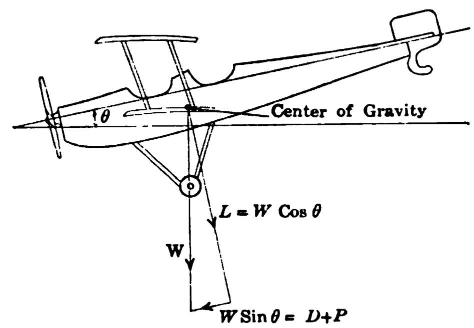

FIG. 4. FORCES ON AN AIRPLANE IN A GLIDE

forward displacement of the machine before landing varies as cos θ and will be a maximum for the smallest value of θ. The pilot has at this angle the greatest radius of action when descending from a height with his engine shut off.

The angle of glide for any machine at any speed can be at once obtained from the total resistance curve for D_t and the weight of the machine, assuming $L = W$ which makes a comparatively small error. In Fig. 1, the angle of glide is shown for all speeds of the machine in question.

References for Part I, Chapter 12

Barnwell's "Aeroplane Design."

"Aeroplane Design," by J. C. Hunsaker, *United States Naval Institute Proceedings*, November-December, 1914.

British Report. 1912-1913. No. 86.

Chassériaud et Espitallier, "Cours d'Aviation."

Chapter XIII

Resistance Computations — Preliminary Wing Selections

Example of Estimate for Parasite Resistance for a British Machine

The B.E.2, mentioned in Chapter 12, will serve as an example of the estimate of the total parasite resistance of a machine. The estimate was arrived at by the Royal Aircraft Factory after the most careful tests, both at the N. P. L. laboratory and in full flight, and is given in Table 1.

TABLE 1.

B.E.2; WEIGHT. 1650 POUNDS; 372 SQUARE FEET BIPLANE SURFACE; 70 HORSEPOWER ENGINE.

Estimated parasite resistance at 60 miles per hour.

Part.	*Whence obtained.*	*Value in pounds per square foot of projected area.*	*Resistance in pounds.*	
Struts				
8, 6′0″ × 1¼″	N.P.L. Test		4.2	
4, 4′0″ × 1¼″	"		1.4	
6, 3′0″ × 1¼″	"		1.6	
				7.2
Wings				
2, 20′ cable	"		29.5	
70, 12 G.H.T. wire	"		5.6	
52 strainers	Estimated		3.0	
				38.1
Rudder and elevators			2.0	
Body with passenger and pilot.	N.P.L. Test		40.0	
Axle			2.0	
Main skids and axle mounting			1.0	
Rear skid			.5	
Wheels	N.P.L. Test		3.5	
Wing, skids, wiring, plates, step, silencers, etc	Estimated		10.0	
				59.0
Exposed to a slip stream of 25 feet per second, for an airplane speed of 60 miles per hour, i. e., 88 feet per second.				
Body			40.	
4, 4′0″ struts			1.4	
2/3 of 3′0″ struts			.8	
50′ cable			6.7	
30′ H.T wire			2.4	
Rudder and elevator			2.0	
Rear skid			.5	
Fittings			2.0	
			55.8	
In slip stream resistance increased to			91.5	
Increase				35.7
Total				140.0

One of the most interesting features of this resistance estimate is the allowance for slip stream. The parts of the airplane included in the slip stream are, of course, taken within the area swept out by the propeller. The speed of the machine is 60 miles per hour, i.e., 88 feet per second, and the slip stream is 25 feet per second, i.e., 28.4 per cent, increasing the resistance of the parts involved by some 65 per cent.

Examples of Parasite Resistance Distribution in School Machines

Table 2 furnishes useful estimates of parasite resistance distribution for a number of standard school machines. The slip stream velocity has been taken as 15 per cent of the airplane speed, giving an increase in resistance of 32 per cent for the parts exposed to it.

Discrepancies in these values arise from a number of causes. The Martin has interplane ailerons, and the other machines have wing flaps. The Curtiss has a water-cooled motor with radiator in front. The machine designed at M. I. T. has a radiator above the upper wing. The Curtiss has a two-wheeled landing gear, while the Martin has a third wheel in front.

TABLE 2.

Percentage of Parasite Resistance.

Machine.	Body radiator hoods, cockpits, windshields, 15% allowance made for slip stream.	Tail, rudder, stabilizer braces, wiring, etc.	Landing gear complete.	Wing bracing, struts, wires and fittings.	Ailerons or wing flaps, pulleys, braces, etc.	Expression for parasite resistance P in lbs., V in m.p.h.	Parasite resistance at 60 m.p.h
Curtis 90 h.p., two place, 1893 lbs. tractor	39.5%	10.5%	17.5%	28.5%	4.0%	$P = .035V^2$	126 lbs.
Martin, 1800 lbs., 70 h.p. Renault	28.8%	18.7%	14.1%	14.7%	22.7%	$P = .042V^2$	151 lbs.
90 h.p. biplane, 1850 lbs., tractor, designed at M.I.T.	36.0%	15.1%	18.8%	26.6%	3.5%	$P = .032V^2$	115 lbs.

Two tractor machines, carefully designed by students at the Massachusetts Institute of Technology, gave the following figures:

Type.	Weight in lbs.	Engine.	Parasite Resistance Coefficient, V in m.p.h.	Allowing 10% increase for slip stream.	Resistance at 60 m.p.h.
Tractor Biplane Reconnaissance.	2300	120 h.p.	$P = .040\ V^2$	$.044\ V^2$	158 lbs.
Tractor Biplane Reconnaissance.	2885	125 h.p.	$P = .0485\ V^2$	$.0530\ V^2$	191 lbs.

Parasite Resistance Coefficient for a Sturtevant Seaplane

For a Sturtevant seaplane, weighing 2650 pounds, with a 140 horsepower engine, and where parasite resistance, on account of the floats, is higher than for a land machine of the same weight, the structural resistance is estimated as being given by the formula $P = .0532V^2$, and 10 per cent increase on all the parasite resistance is allowed for, bringing up the value to $P = .0576V^2$, or 212 pounds approximately, at 60 miles per hour.

Allowance for Slip Stream

The question of slip stream velocity is one of great complexity, and, in the present state of knowledge, it does not seem advisable to enter into very complicated calculations when working out performance curves. The estimated figures given for the various American machines seem to be very well borne out by tests in the field. The British allowance for slip stream increase was 28.4 per cent, and the one given by American practice is 15 per cent. It would be safe to say that if for the parts of the machine, within the area swept out by the propeller, the speed is increased by some 20 per cent, and resistance of those parts increased by some 44 per cent, a sufficiently accurate estimate will be made.

The other method adopted of increasing the total structural resistance by some 10 per cent to allow for slip resistance, though not so rational, has the advantage of being simpler, and is still in accordance with tests in the field. For a monoplane, where the parts exposed to the slip stream bear a larger ratio to the rest of the machine, an increase of 15 per cent on the total structural resistance is probably advisable.

Preliminary Estimates for Parasite Resistance

In making preliminary estimates for a machine, a really difficult point is the allowance to be made for parasite resistance. Some authorities allow for the parasite resistance by finding the resistance of the body and multiplying it by four for a biplane and by three for a monoplane. Such rules can only be roughly correct, and it is best to refer to data for standard machines and select parasite resistance coefficients of a machine of similar type and weight. The figures given in this section will be sufficiently accurate for a preliminary design.

Preliminary Selection of Wing Section and Area

A great many ingenious methods have been devised for the selection of correct wing sections and areas for the preliminary design of a machine whose engine-power and specification are given. Eiffel, among others, has developed a very complete system. It seems best, however, to employ the simplest and most straightforward trial and error methods, based on the following rules:

(1) From a consideration of standard practice, select the loading per horsepower and hence weight of the machine.

(2) From a consideration of standard practice, select the approximate loading per square foot.

(3) From some such considerations as those given in Chapter IV select two or three wings which are likely to give the qualities desired.

(4) Assume a parasite resistance coefficient which from a standard practice is likely to apply to a machine of the type and weight in question.

(5) Draw up a number of performance curves varying:
   (a) Wing sections
   (b) Area for each wing section
   (c) Assumed propeller efficiency curves.

Some data on standard practice will be given in the Second Part of the book, and the above rules will be applied to the design of a standard machine.

References for Part I, Chapter 13

Barnwell's "Aeroplane Design."
British Report, 1912-1913. No. 86.

Part II

Airplane Design

Chapter I

Classification of Main Data for Modern Airplanes Unarmed Land Reconnaissance Machines Land Training Machines

The Army Classification

Constructors in America have hitherto mainly developed one type of airplane, the tractor biplane reconnaissance machine. But with the rapid development of military aeronautics, airplanes are evolving into distinct classes, just as the component vessels of a fleet. The memorandum on "Military Airplanes," prepared by the office of the Aviation Section of the Signal Corps, offers the most authoritative classification, and one which constructors must of necessity follow very closely. It suggests six distinct types, which we shall study as closely as possible, within the limits of data held confidential by manufacturers. (I) Land Reconnaissance Machine, used when there are no enemy airplanes; (II) Land Primary School Machine; (III) Land Advanced School Machine; (IV) Land Gun Carrying Machine, (V) All-round Twin-engined, Land or Water, (VI) Land Pursuit Type.

Unarmed Land Reconnaissance Machine

For this machine, the memorandum gives the following figures:

TABLE 1.

	Unarmed Land Reconnaissance Machine.
Horsepower	130
Pusher or tractor	Tractor
Number of men	2
Military load, pounds	475
Fuel load, pounds	450
Miles radius of action, full power	415
Climb in 10 minutes, feet	3,400
High speed, miles per hour	82
Low speed, miles per hour	46
Factor of safety	7
Percentage made in war	2

(Gross load is well over 2,500 pounds for this type.)

The memorandum deals very unfavorably with this type of machine, which forms, as we have said, an important part of American construction. It is said to be a false development, suitable only for use against an enemy who has no airplanes. Possibly useful for long-range reconnaissance, it will be outmatched in warfare by the armed "pursuit" type and the large armed twin-engine machine. For short ranges the pursuit type will surpass it, for long ranges the larger machine may be not quite so rapid, but will have a greater radius of action. A careful study of these views would lead one to believe them correct and in accordance with developments abroad.

Analysis of Main Data for Representative Unarmed Reconnaissance Biplanes More Than 2500 Pounds Gross Weight

In Table 2 are given the main dimensions and performances of a number of representative machines. Such analysis and the empirical rules to be derived from it are invaluable in the preliminary stages of a design, and enable the designer to avoid misleading rough estimates of weights and dimensions.

TABLE 2.

MAIN DATA FOR TWO-SEATER TRACTOR BIPLANES OF THE UNARMED RECONNAISSANCE TYPE OVER 2,500 POUNDS IN WEIGHT. RECENT EXAMPLES OF CONSTRUCTION.

Machine	Standard H-3	Curtiss R-4	Wright-Martin V	Sturtevant S	Wright-Martin R
Engine	Hall-Scott A-5	Curtiss	Hispano-Suiza	Sturtevant	Hall-Scott A5a
Horsepower	135	200	150	140	150
Number of cylinders	6	8	8	8	6
Revolutions per min.	1,250	1,400	1,450	2,000	1,375
Gasoline tank capacity	68 gallons	100 gallons	...	...	70 gallons
Endurance in hours	6	5.42	appr. 6	4.5	4.84
Maximum speed, miles per hour	84	90	...	86	86
Minimum speed, miles per hour	46	48	...	42	47
Climb in 10 minutes, feet	3,400	4,000	...	3,500	3,500
Propeller diameter (two blades)	9'	...	8' 4"	...	8' 6"
Weight loaded, pounds	2,700	3,245	2,310	2,550	2,880
Weight bare, pounds	1,900	2,225	1,725	1,850	1,905
Useful load, pounds	800	1,020	905	700	983
Percent useful load	29.6	31.4	34.2	27.4	34.2
Weight per horsepower in pounds	20.0	16.21	16.86	18.2	19.1
Weight per square foot wing area in pounds	5.08	6.42	5.86	4.64	6.25
Overall length	27' 0"	29' 0"	27' 2"	27' 0"	26' 8"
Mean span of wing/length	1.41	1.49	1.37	1.65	1.64
Wing section	R. A. F. 6	R. A. F. 6	Vought 4	R. A. F. 6	...
Upper span	40' 1"	48' 4¼"	39' 8½"	49' 6"	50' 8"
Upper chord	6' 6"	6' 3"	5' 9"	6' 3"	5' 6"
Upper aspect ratio	6.2	7.75	6.95	7.95	9.25
Lower span	40' 1"	38' 5⅛"	39' 8½"	39' 6"	36' 10"
Lower chord	6' 6"	6' 3"	5' 9"	6' 3"	5' 6"
Lower aspect ratio	6.2	6.15	6.95	6.32	6.6
Gap	6' 6"	6' 2"	5' 7"	6' 3"	6' 0"
Gap, lower chord	1.00	0.98	0.98	1.00	1.09
Total area of wings, including ailerons, in square feet	532	505	430	540	458
Area of rudder in square feet	10	16.5	12.37	appr. 15	8.7
Area of vertical fin in square feet	5	7	5 appr.	...	7.3
Area of elevator in square feet	23	27.5	...	24	...
Area of stabilizer in square feet	32	40.5	51.2 (elevator and stabilizer)	28	53.2 (elevator and stabilizer)
Ailerons upper wing	31	33.8	32.3	39	48
Ailerons lower wing	31	20.5	32.3	39	...
Type of fuselage	Rectangular	Rectangular	Rectangular	Triangular	...
Dihedral	3°	3°	1° 15'	2°	1°
Stagger	10°	5° appr.	1 ft.	None	20.4% chord length
Sweepback	10°	None	None	None	None

Average Values for Machines Over 2500 Pounds in Weight

This group is composed of excellent, controllable machines very similar in character. It is therefore possible to draw some fairly definite conclusions.

(a) Average gross weight, 2737 pounds.

(b) Average wing area, 493 square feet.

(c) Average horsepower, 155. The latter figure is considerably increased by the inclusion of the Curtiss R-4 with its 200 horsepower engine. There is a tendency to give higher power to this class, with correspondingly better performances.

(d) Average endurance, 5.65 hours. This figure is probably a very fair value of the endurance possible if good climb is to be maintained. It should be noted that it would be possible to take up much more fuel, and not decrease the speed; in fact, to increase it slightly. At the same time, minimum speed would be increased.

(e) Average weight per horsepower, 18.1 pounds. The Curtiss lowers this average value, and is an indication of what will follow when lighter new engines, such as the new Thomas and Sturtevant, enter into construction.

(f) Average weight per square foot of wing area, 5.65 pounds.

(g) Average maximum speed, 86.50 miles per hour.
Average minimum speed, 45.75 miles per hour.
Average climb in 10 minutes, 3666 feet.

The number of machines considered is too small for curves to be plotted, but it is interesting to see how in diminishing the weight per horsepower from 24.2 pounds to 16.21 pounds the maximum speed increases from 84 to 90 miles per hour, while the low Sturtevant wing loading gives a landing speed of 42 miles per hour as compared with the Curtiss of 50 miles per hour.

(h) Average of ratios of $\frac{\text{mean span}}{\text{overall length}} = 1.51$.

This is an important point to be considered in the design of a machine. As we shall see later in considering longitudinal stability, it is quite possible to secure adequate static stability by using a short body with a large tail surface placed at a negative angle. But an excessively short body, although it means saving in weight, may fail to give dynamic stability, due to lack of damping. At this stage of the science, we can only fix on a length for the body by taking average values such as the above.

(i) Average aspect ratio upper wing, 7.60.
Average aspect ratio lower wing, 6.76.

There seems in the light of these figures no reason why an aspect ratio of 7.5 for the upper span, and 7.0 for the lower should not be successfully employed.

(j) Gap/chord ratio is practically 1.00 in every case. Without undue conservatism, it would appear that for machines of this size, the increased structural weight of a larger gap/chord ratio is prohibitive, whereas in smaller machines with smaller chord, much greater values might be employed to advantage.

The dimensions of control and stabilizing surfaces present an exceedingly complex problem, so many factors being involved. They will be carefully studied in our design, but in the preliminary stages some of the following empirical relationships may be useful:

(k) Aileron or wing flap area: The dimensions of these will depend on the area of the wings whose rolling moment it may be necessary to overcome, on the weight and lateral radius of gyration of the machine and on the span of the wings which gives the moment arm of the ailerons. These factors are too complex, however, and at present the following formula offers a fairly satisfactory standard of comparison:

$(S_1a_1 + S_2a_2) = CA$, where A = area of wings, S_1 and S_2 = spans of upper and lower wings, a_1 and a_2 = aileron areas on upper and lower wings. C = a constant. Where C is large there is powerful lateral control, where C is small there is weak lateral control. The values for the above five machines are as follows:

Standard H-3	Curtiss R-4	Wright-Martin V	Sturtevant S	Wright-Martin R
4.65	4.50	5.95	6.50	5.33

Too powerful lateral controls present difficulties in handling just as too weak controls. The average value of $C = 5.38$ might be at least some guide.

(l) For the horizontal elevator and stabilizer, the following very rough formula is sometimes employed in preliminary work, based on ideas similar to those enunciated in the previous paragraph:

$d = \frac{QL}{AC}$, where d = some constant, Q = area of elevator and stabilizer, L = overall length, A = area of wings and C = mean chord.

The following constants hold for our five well controlled machines:

Standard H-3	Curtiss R-4	Wright-Martin V	Sturtevant S	Wright-Martin R
.429	.625	.507	.416	.561

These constants are fairly close together, with an average value of .507. A big value of d means powerful control. Without further analysis, it is seen from Table 1 that the stabilizer is made between 20 to 50 per cent larger than the elevator.

(m) Similarly for vertical surfaces, if $f = \frac{VL}{AS}$, where f = constant, V = vertical area of rudder and fin, L = length, A = area of wings and s = mean span of wings, we find

Standard H-3	Curtiss R-4	Wright-Martin V	Sturtevant S	Wright-Martin R
.019	.031	.027	.015	.022

Average value, .023.

We shall discuss the problem of vertical fin and rudder area more closely later.

Primary and Advanced Training Airplanes

In the training of military pilots similar methods are now employed in the majority of schools, and there are two distinct stages, "primary" and "advanced" training.

On the primary machine, the aviator obtains his first certificate, and the requirements of this type tend toward a steady, slow type of machine, in which it is easy to acquire confidence. The advanced training machine is scarcely distinguishable from the land reconnaissance machine, although it is somewhat slower. In the memorandum on Military Airplanes, the following suggestions are made for these two types, which are of obvious and permanent utility.

TABLE 3.

	Land Primary School, can also be used for field artillery fire control	Land Advanced School, may possibly be used for mountain and forest tactical reconnaissance.
Horsepower	80	100
Pusher or tractor	Tractor	Tractor
Number of men	2	2
Military load, pounds	375	400
Fuel load, pounds	150	240
Miles radius of action, full power	195	300
Climb, feet in 10 minutes	2,000	3,000
High speed, miles per hour	66	75
Low speed, miles per hour	37	43
Factor of safety	7.5	7.5
Percentage made in war	25	20

In designing training machines, the constructor has the advantage of complete specifications issued by the Signal Corps (Aeronautical Specifications, Nos. 1001 and 1002). These specifications are readily obtainable, but some of the main points are set forth here, as they will be applicable to our design of a standard machine, and must be constantly kept in mind by the designer.

MODERN AMERICAN TWO-PLACE TRACTORS

These Photographs Show Representative Two-Seater Tractor Biplanes of the Unarmed Reconnaissance Type, Weighing Over 2,500 Pounds

THE WRIGHT-MARTIN, MODEL R, TRACTOR

STURTEVANT S TRACTOR

MODEL V. WRIGHT-MARTIN TRACTOR

THE STANDARD, MODEL H-3, TRACTOR

THE CURTISS R-4 MILITARY TRACTOR

IMPORTANT POINTS IN SPECIFICATIONS NOS. 1001 AND 1002 FOR MILITARY TRAINING AIRPLANES

Primary	*Advanced*
1. Tractor biplane, useful load:	
(a) Pilot and passenger. 330 pounds. Three hours	(a) Pilot and passenger, 330 pounds. Four hours
(b) Gasolene, oil and water.	(b) Gasolene, oil and water.

2. Curtiss eight-cylinder, OX-2. 90 horsepower at 1400 revolutions per minute, or an approved American made engine between 70 and 90 horsepower, for the primary, and between 90 and 110 horsepower for the advanced.

3. Minimum speed, 37 miles per hour.	Minimum speed, 43 miles per hour.
Maximum speed, not less than 66 miles per hour.	Maximum speed, not less than 75 miles per hour.
4. Fully loaded machine must attain an altitude of 10,000 feet in	
Two hours.	75 minutes.
5. Climb in 10 minutes shall be not less than	
2600 feet.	3000 feet.

6. Celerity of response to control, the proper degree of symmetric and assymmetric stability (static and dynamic); steadiness in disturbed air, etc. Satisfactory manoeuvering on the ground.

7. Both the dual Curtiss (shoulder or chest yoke) and dual Deperdussin types of control ready for installation in cockpits.

8. Factors of safety.

(a) Main plane structure. Conditions assumed:

(1) Load as above.

(2) Angle of incidence of mean chord of main planes: that of maximum lift coefficient.

(3) Air speed: that normally corresponding to the above load, and angle of incidence for the net effective surface area.

Factor of safety not less than 7.5.

(b) Body and tail structure.

(1) Air speed, 100 miles per hour.

(2) Angle of incidence of fixed horizontal tail surface, minus 6 degrees; elevator surface minus 20 degrees.

Factor of safety not less than 2.5.

9. A complete outfit of instruments, tools, pressure gauges, etc., is specified.

10. Three-wheel type landing gear, the third wheel being 20 x 4 inches, just in rear of the plane of rotation of the propeller; normally not touching the ground, but designed to touch the ground when the mean chord of the main planes is horizontal. Main wheels, 26 x 4 inch tires, and 6 x 1½ inch hubs, with spokes.	Landing gear of two-wheel type. Wheels, 26 x 4 inch tires, and 6 x 1½ inch hubs.

11. Body shall be of one part, not the jointed tail type. All turnbuckles in the body wiring to be readily accessible. In the side wiring they should be near the upper longerons. The wing spar fittings on the body to which the lower planes are attached shall be tied together across through the body by steel tubing. The interior of the body shall be so constructed as to permit thorough inspection of all wiring, control leads, etc. As far as practicable all leads shall be direct.

12. The design and mounting of the tail skid and vertical rudder shall be such as to prevent injury to the vertical rudder in case of failure of the tail skid.

13. The number of different sizes of turnbuckles shall be reduced to the minimum. Pulleys, pins, bolts, turnbuckles, etc., drilled for safetying. Safety wire shall be of No. 18 gage copper wire.

14. Satisfactory fields of vision.

15. Seats in tandem, padded cockpits, safety belts.

16. Housing around power plant readily detachable. Convenient access to all parts of the engine which may require adjustment or inspection.

17. Radiator proof against vibration.

18. Gravity feed throughout preferred. A positive and reliable system of pumping may be used; in which case a gravity feed tank holding at least forty minutes supply to be embodied.

19. Upper plane to extend beyond the lower plane laterally by an amount approximately equal to the chord. Lateral control to be by means of trailing edge flaps on the upper plane only.

20. Stranded steel cable shall be used for all tension members which are readily accessible for adjustment and for all control leads. Structural tension members shall be of hard cable, and control leads shall be used for terminals of hard, single-strand wire. No spliced terminals in hard cable will be accepted. All cables which are members of the wing structure and normally under tensile load in flight shall be in duplicate and made independent between fittings. Satisfactory provision for convenient and thorough inspection of control cables and pulleys and vital structural members. In the internal wing bracing, the compression members carrying the drag of the wings shall be separate wooden struts and not wing ribs. Rib webs shall be reinforced between lightening holes to strengthen them in longitudinal shear.

The above specification not only provides an excellent guide for the design of the school machine, but is also a guide to the performance which may be expected of this type. A machine might be perfectly acceptable, however, even if it did not adhere rigidly to the above specification, provided its main requirements were successfully carried out. Particularly is this true as regards the engine power.

Data for a Typical School Machine Less Than 2000 Pounds in Weight

The Curtiss JN4-B, for which full data is supplied by the manufacturer is a good example of this type, and in default of a classification such as that of Table 2, should prove a reliable guide in preliminary design.

CURTISS JN 4-B

Engine, Curtiss OX; horsepower, 90; cylinders, 8; revolutions per minutes, 1400; weight per rated horsepower, 4.02 pounds; bore and stroke, 4 x 5 inches; fuel consumption per hour, 9 gallons; fuel tank capacity, 20 gallons; oil capacity, 4 gallons; fuel consumption per brake horsepower per hour, 0.60 pounds; oil consumption per brake horsepower per hour, 0.030 pounds.

Maximum speed, 75 miles per hour; minimum speed, 43 miles per hour; climbing speed, 3000 feet in 10 minutes.

Net weight, machine, empty, 1320 pounds; gross weight of machine and useful load (fuel for 4.16 hours), 1905 pounds; distributed as follows: 225 pounds fuel, 30 pounds oil, 165 pounds pilot, 165 pounds passenger. Total, 585 pounds. Percentage useful load, 30.7 per cent.

Total supporting surface, 356.7 square feet; loading per brake horsepower, 21.16 pounds; loading per square foot of supporting surface, 5.3 pounds.

Wing section, Eiffel, 36; upper span, 43 feet 7⅜ inches; upper chord, 4 feet 11½ inches; lower span, 33 feet 11¼ inches; lower chord, 4 feet 11½ inches; gap, 5 feet 2 3-16 inches; overall length of machine, 27 feet 3 inches; overall height, 9 feet 10½ inches; ratio of mean span to overall length, 1.43.

Dihedral, 2½ degrees; sweepback, .0 degrees; stagger, 12 5-16 inches.

Control surfaces.—Ailerons (upper wing), 35.28 square feet.

Constant in formula $(S_1 a_1 + S_2 a_2) = CA$. $C = 4.3$. Horizontal stabilizer 28.7 square feet; elevator 22.0 square feet. Constant in formula, $d = \frac{QL}{AC} d = .766$. Rudder 12.00 square feet; Vertical fin 3.80 square feet. Constant in formula, $f = \frac{VL}{AS}$, $f = .029$.

Photographs and Drawings

Considerations of space do not permit inclusions of drawings of these types. The accompanying photographs are representative. Details of construction and drawings will be fully developed in our design in subsequent sections, and drawings of one or two representative machines will precede this design.

In the ensuing chapter, we shall study the pursuit type, the Twin Engine Machine, and the armed reconnaissance type.

References for Part II, Chapter 1

"Memorandum on Military Airplanes," Prepared in the office of the Officer in Charge of the Aviation Section, Signal Corps, U. S. A. AVIATION AND AERONAUTICAL ENGINEERING, September 15, 1916.

"Aeronautics," by J. C. Hunsaker, in the *Mechanical Engineer Handbook*.

EXAMPLES OF PURSUIT TYPE AIRPLANES

German Albatross of 1916

The Curtiss Triplane

Photo. from Underwood and Underwood

A Nieuport Pursuit Machine

EXAMPLES OF GERMAN ARMED BIPLANES

An L. V. G. of 1916

An Aviatik Gun-Carrier

Chapter II

Land Pursuit Machine
Land Gun Carrying Machine
Twin Engined All 'Round Machine

The High Speed Scout or Land Pursuit Type

The high speed scout or pursuit type has in the present war assumed a very great importance. In the War Department memorandum on "Military Airplanes," its functions are well defined:

"By virtue of its tremendous speed and climbing ability, it can dodge and outmaneuver its larger enemy, maintaining an effective fire with its machine gun, at the same time presenting a small and bewildering target. This is an ideal machine for tactical reconnaissance. It can even drop a few bombs where they will do most good."

The United States Army memorandum gives the following figures pertaining to this type:

TABLE 1.

LAND PURSUIT TYPE.

Horsepower	110
Type	Tractor
Number of men	1
Military load, pounds	200
Fuel load, pounds	150
Miles, radius of action, full power	315
Climb, feet in 10 minutes	8000
High speed, miles per hour	115
Low speed, miles per hour	43
Factor of safety	7.5
Percentage demand in war	21.0

Very few machines of this type have been built in America. Abroad such machines have been used in great numbers, but little information is available for recent French and English types, such as the Nieuport, Morane, Vickers, Bristol, Sopwith, etc., with light rotary engines of between 80 and 130 horsepower. Lately very light and more powerful 150 horsepower V-type Hispano-Suiza engines have been employed in great numbers.

We may say that an average of 120 horsepower is used in this type, that it is a single seater machine, almost always a biplane with the smallest possible wing spread, a tractor with a light machine gun firing either through the propeller or above the wings.

English opinion based on experience in the war supports an inherently stable machine which the pilot can leave uncontrolled for a short period while engaged in combat or other functions. To obtain inherent stability in this type is a difficult problem. The high loading per square foot of area is not conducive to stability, and the employment of correct fin areas and dihedrals is still a problem. The high loading also introduces difficulties from the point of view of stresses in the wing structure. Nothing lower than 7 pounds per square foot of wing area seems possible.

With the production in the United States of such engines as the General Vehicle Company's Gnome and the Hispano-Suiza, there is to be expected a very rapid increase in the number of American speed scouts. These light and powerful engines will enable the weight per horsepower to be diminished and the speed and climb to be increased until European practice is equaled.

Data for Pursuit Type, 100 Horsepower Engine

In Table 2 some data has been collected bearing upon some of these types. Little detailed information is available, but these figures and illustrations should be sufficient to give a general idea of present development.

TABLE 2.

DATA FOR HIGH-SPEED PURSUIT MACHINES.

Model	Nieuport	S.P.A.D.	Bleriot	Curtiss triplane
Engine	80 h.p. Le Rhone	150 h.p. Hispano-Suiza	150 h.p. Hispano-Suiza	OXX-2 100 h.p.
Number of men	1	1	1	
Endurance at full speed (hours)	2½ to 3	..	..	2½ to 3
Maximum speed (miles per hour)	..	125	125	120
Minimum speed (miles per hour)	..	..	..	56
Climb in 10 minutes (feet)	6,000-7,000	9,200	9,200	10,000
Climb to 3,200 feet (minutes)	..	3	3	..
Climb to 6,400 feet (minutes)	..	6	6	..
Climb to 9,600 feet (minutes)	..	10½	10½	..
Total weight (pounds)	..	..	..	1,218
Useful load	..	460	460	..
Pounds per horsepower	..	..	..	12.18
Area wings (square feet)	..	185	185	143
Wing loading (pounds per square foot)	..	..	..	8.5
Wing area per brake horsepower (square feet)	..	1.23	1.23	1.43
Span top plane	24' 6"	..	..	25' 20" all three planes
Chord top plane	3' 11"	..	..	2'0" all three planes
Aspect Ratio top plane	6.15	..	..	12.5 all three planes
Span bottom plane	23' 0"	..	..	..
Chord bottom plane	2' 4"	..	..	..
Aspect ratio bottom plane	9.9	..	..	..

Data for More Powerful Pursuit Types

So rapid is the development of the foreign military airplanes that it would seem as if the speed scout fitted with a rotary engine of about 100 horsepower, is now being displaced by a more powerful type.

It is interesting to note that in the S. P. A. D. and Bleriot types the climb does not apparently fall off with altitude, showing probably that the difficulties of maintaining the power of the engine at high altitudes have been successfully met.

Trend of Design in the Pursuit Type

Among the salient features of this type is the very small weight per horsepower, 12 pounds or thereabouts, as compared with the 18 or 20 of the large reconnaissance type. There is a tendency to employ large aspect ratios—the small weight per horsepower and small wing surface permitting the chord to be cut down considerably. Thus we find in the Nieuport aspect ratios of 6.15 and 9.9; in the Curtiss triplane 12.5 on all the planes. In the triplane the extraordinary aspect ratio of 12.5 is rendered possible by the distribution of the carrying surface among three planes instead of two. Stream-lining to the very limit is another feature.

Hitherto German constructors do not appear to have sought the reduction of head resistance to any great extent, but the new Albatross shows a beautiful body, struts reduced to a minimum, and the stream-lining carried to the extent of a hemispherical nose-piece over the propeller boss. In the Nieuport scout the V-strut system gives both lightness of construction and aerodynamic efficiency. The wonderful improvement of the Curtiss triplane with no larger power, as compared with its predecessor, the "Baby" scout, is due in part to the large aspect ratio which counterbalances the inefficiency of the triplane, but more to the clever *K* strut construction with the two tubular struts stream-lined in, and the stream-lined chassis construction. Such valuable yet simple ideas in the design of this type are well worth attention.

German practice in every respect seems to be following French practice very closely for this type. Recent information at hand shows that the small biplane, such as the Albatross just mentioned and the new Fokker biplane are being built in great numbers. The Fokker biplane has apparently superseded the monoplane.

Guns on the Pursuit Type

However stable a machine may be, and even if it is equipped with a stabilizer or automatic pilot, it seems impossible that a pilot should at the same time be a gunner capable of firing in all directions. The light machine guns which are employed, are probably fired straight ahead towards the enemy machine which the pilot is approaching. They may be

(1) fired through the propeller, which is suitably protected for deflecting stray bullets.
(2) fired through the propeller with a suitable synchronizing mechanism as on the Fokker (see appended reference).
(3) placed above the wing within reach of the pilot as on the Nieuport.

Land Gun Carrying Machines

For the land gun-carrying type of airplane the War Department memorandum specifies:

TABLE 3.

Horsepower	130
Type	Tractor
Number of men	2
Military load, pounds	500
Fuel load, pounds	425
Miles radius of action, full power	360
Climb, feet in 10 minutes	3100
High speed, miles per hour	77
Low speed, miles per hour	45
Factor of safety	7.0
Percentage demand in war	4%

If any criticism is to be ventured with regard to the memorandum, it is on the score of too implicit faith in the all round twin-engined machine, and neglect of the importance of the gun-carrying machine of this type. Most of our information as to this type comes from the splendid descriptions of captured German machines in *L'Aérophile*. The Germans have succeeded in providing this type with one or two machine guns, with a good range of fire and arrangements for throwing bombs, and yet with excellent performance.

TABLE 4.

DATA FOR GERMAN TWO-SEATER GUN-CARRYING MACHINE (From *L'Aérophile*).

Machine	Rumpler 1916	Aviatik 1916	L. V. G. 1916
Engine	Mercedes 165 horsepower	Mercedes 170 horsepower	Mercedes 165-170 horsepower
Maximum speed on ground, miles per hour	92	93	95
Minimum speed on ground, miles per hour	..	49.5	..
Speed at 1,000 meters, miles per hour	..	84	87
Speed at 2,000 meters, miles per hour	85	82	84
Speed at 3,000 meters, miles per hour	81.5	82	..
Climb, in feet per minute	3,200 in 10 6,400 in 16 9,600 in 29	1,600 in 4½ 3,200 in 9½ 6,400 in 21½ 9,600 in 47½	3,200 in 12 6,400 in 33
Maximum altitude fully loaded, feet	14,800	11,200	10,000
Endurance, fully loaded, hours	4	4½	4½
Total weight, pounds	2,780	2,840	2,840
Net weight, pounds	1,830	1,860	1,860
Useful load, pounds	950	980	980
Percentage useful load	34.2%	34.4%	34.4%
Military load, pounds	580	550	550
Loading per brake horsepower, pounds	16.8	16.7	16.8
Loading per square foot of wing area, pounds	7.35	6.6	7.1
Overall length, feet	25.2	..	26
Total wing area, square feet	378	430.56	400
Upper span, feet	39.2	40.66	41.3
Chord, feet	5.4	6.1	..
Lower span, feet	35.4	35.4	36.4
Chord, feet	5.4	6.1	..
Gap, feet	..	6.3	..
Ailerons, square feet	27.8	..	22
Stabilizer area, square feet	35.4	35.2	37.6
Angle of stabilizer	— 1.36°	— 1°	..
Elevator area, square feet	18.2	12.07	16.2
Fixed vertical fin, square feet	4.8	3.2	5.3
Rudder, square feet	7.25	6.4	6.4
Dihedral of wings	2°	Slight	None
Angle of wings to body	5°	4.5°	4.25°
Décalage	None	Almost none	None
Stagger	None	Almost none	None
Sweepback	Slight	None	Slight
Armament	Pilot forward machine gun firing through propeller. Passenger in rear with circular gun-turret. Some of these machines also carry a bomb dropping device.	In rear seat pilot has two bomb-throwers in front of him. Passenger in forward seat has two adjustable gun supports, permitting him to fire in almost all directions. The propeller is protected by a circular plate.	Complicated arrangement for throwing bombs which sometimes weighs as much as 25 pounds. Two machine guns, one firing through the propeller, the other on a rotary turret in rear cockpit.

In Table 4 some useful data has been collected, and photographs of the Aviatik and L. V. G. shown herewith, are good examples of this type. Recent development in America has shown the necessity of standardized tests at various altitudes and it is interesting to note that figures are given for speeds on the ground and at various standard altitudes; while maximum altitude, or "plafond," as the French term it, is also specified.

These German machines, as regards climb, are on a par with the American unarmed reconnaissance machines. They appear to be much superior in maximum speed. Their percentage of useful load is greater. By careful construction and probably with a lower factor of safety, they actually have a lower loading per brake-horsepower, even when carrying two machine guns, the ammunition required, and bombs and bombing devices. This indicates how much room there still is for improvement in American machines as regards weight saving.

They are short machines, with large stabilizing planes, but comparatively small control surfaces. The wings are very

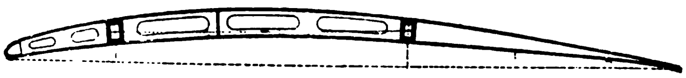

FIG. 1. AVIATIK WING SECTION

heavily cambered, as shown in Fig. 1, yet look as if they were efficient, and deserve careful study. They appear to be an improvement on the wing sections commonly employed. The wings also appear to be set at large angles to the body, so that the propeller axis is coincident with the line of flight at normal angle of incidence, and at a small angle to the line of flight in climb.

Their design is characterized by a robust lightness due to careful utilization of material throughout, and probably a uniform factor of safety.

The Mercedes engine so largely employed on German airplanes has the reputation of being extremely reliable and efficient, but is considerably heavier per brake-horsepower than American engines of about the same power, an important fact to be considered when the American designer sets out to equal the German machine.

Twin-Engined Machines

The memorandum prepared in the office of the officer in charge of the Aviation Section, Signal Corps, U. S. A., gives very interesting data on twin-engined seaplanes, reproduced in Table 5 herewith. It lays strong stress on the advantages of this type, partly on the ground that a thoroughly developed engine of over 200 horsepower had not yet been constructed; but since that time progress has been made and the 300 horsepower engine is becoming a practical possibility. Stress is laid on the versatility of the twin-engined machine. In a land twin-engined machine, with the engines supported on the wings, and a central body, it is suggested that the pilot with his controls could be placed in the rear cockpit in rear of the propellers; the observer with his machine gun, in the forward cockpit forward of the propellers; and the bombs and gasoline between the pilot and observer, near the center of gravity of the complete system. With such an arrangement, the observer would have an ideal field for observation and for gun fire; with a stabilizer the pilot could fight the machine to the rear, and the machine would be an excellent fighting airplane as well as a bomber. By decreasing the bomb weight, the radius could be increased to a very long range. A third man could be installed in between the front and rear cockpits by making slight alterations in the construction of the body. If the third man had charge of the controls, there would be an even better fighting machine. If neither bombs nor a third man were carried, the landing wheels could be replaced by two or three pontoons, giving a military seaplane.

TABLE 5.

Horsepower	260
Type	Pusher or tractor
Number of men	2
Military load, pounds	550 to 1100
Fuel load, pounds	600 to 1150
Miles radius of action, full power	450 to 600
Climb, feet in 10 minutes	3400
High speed, miles per hour	90
Low speed, miles per hour	47
Factor of safety	7.0
Percentage made in war	4

A further evident advantage of this type is that it could be flown with one engine out of commission. The propeller would not be required to take the great power of a 300 horsepower engine. Although the support of the engine on the wing presents considerable difficulties, yet this arrangement might react favorably on the weight of the wing structure by distributing the load, and therefore giving less bending moment to the inner wing panels.

If a single engine of 250 or 300 horsepower were used, the same military and useful loads would be available. With a single body the parasite resistance would probably be diminished. But the difficulties of constructing such a machine for fighting as well as bombing purposes would be very considerable. With the engine in front, it might be possible for the pilot to sit in the forward cockpit and shoot through the propeller, while in the rear the observer could operate a machine gun in all directions. If bombs and bombing apparatus were included in the single body, there would be an extremely difficult construction problem. If the high-powered machine were made a pusher, the rear occupant would be under difficulties as a machine-gun operator. The other possible alternative would be to place the observer in a cockpit forward of the propeller—which would revolve round the body—the engine behind the propeller, then the pilot in the rear seat. None of these arrangements seem to have the straight forwardness and simplicity of the twin engined type.

It would seem, therefore, that in spite of the development of a reliable 300 horsepower engine, the twin-engined machine would have much to recommend it.

The following points may be disadvantages of the twin-engined machine, or merely problems which careful design can overcome:

(1) There are difficulties in preliminary design. In the school machines, the armed and unarmed reconnaissance types, and in the speed scout, we have data to draw upon from which loading per square foot of wing area, loading per brake-horsepower, useful load, etc., can be at once fixed within certain limits. Here we have an entirely new problem. Theoretical considerations show that with increased span, the bending moment and other stress producing forces for geometrically similar machines vary as the cube of the span. The resistance of bracing wires would vary as the square of the span. The section modulus of the wing spars would vary as the span cubed, but their area, resisting direct tension or compression would only vary as the square of the span. Similar considerations would follow for other strength members. Even if we allow the structural weight advantage of engines on the wings as previously mentioned, the conservative designer would still expect weights to go against him. If the twin seaplane that he is designing follows the same outlines of construction that he has been accustomed to in a single-engined type, he should allow for a slightly heavier loading per horsepower and a slightly smaller loading per square foot of wing area, i. e., a larger wing area than he would, in the light of past experience, expect to employ. Such a conservative outlook—well founded in the author's opinion, particularly for experimental machines—would lead again to less favorable estimates of performance, and avoid the ridiculously optimistic estimates for these large machines in the recent bids. It is interesting, in the light of these remarks, to study the performances submitted in these

bids for seaplanes. (AVIATION AND AERONAUTICAL ENGINEERING, December 1, 1916.) The Curtiss Company, with its past experience in the building of twin-engined machines (Twin JN), specified a climb of 2,000 feet and a speed of 65 miles per hour, although probably equipped with two 200 horsepower engines. The Wright-Martin Corporation was so conservative as to specify no performance. A great many optimistic bids were submitted. The average climb sent in appears to be 3,580 feet, the average maximum speed 77 miles per hour. One would be inclined to think that with present American methods of construction, a climb of 2,800 feet, and a maximum speed of about 68 miles per hour would be as much as could possibly be expected.

(2) Control surfaces and stability furnish interesting problems. The effect of placing the engines out on the wings is to increase the moment of inertia in roll, and it is very difficult to say what effect this will have on inherent lateral stability. It is certain that the aileron area required will be somewhat greater than that in a machine where the engine is at the center of gravity and that the machine will be slow to respond to lateral controls. The moment of inertia in yaws will similarly be increased so that rudder and vertical fin surfaces may have to be larger proportionately than on the usual machine. These are points requiring the most careful attention in design.

(3) Another problem in connection with the twin-engined machine is that of propeller slip from the two screws, both turning inwards. This symmetrical arrangement is prescribed by the Army specifications as avoiding torque and gyroscopic effects. The down stream from the propellers impinging on the stabilizer is said to increase the safety from the point of view of longitudinal balance, giving tail heaviness with power, and nose heaviness without power. The exact effects are, however, still open to experimentation.

Space forbids a discussion of numerous other points which this type presents. The appended references will give the reader some information. The German twin-hydro reproduced in AVIATION AND AERONAUTICAL ENGINEERING, September 15, 1916, is of particularly neat construction, the specification No. 1002 is almost a text-book on design, and the S. A. E. paper on twin-engined machines read by Lieut. Col. V. E. Clark, U. S. A., touches on a greater number of points than we are in a position to deal with. Anyone setting out to construct such a type would do well to devote considerable time to wind tunnel experimentation, computation of moments of inertia, etc.

References for Part II, Chapter 2

"GUN-CARRYING TWO-SEATER MACHINES."

"Rumpler," *L'Aérophile*, December, 1916.
"Aviatik," *L'Aérophile*, October, 1916.
"L. V. G.," *L'Aérophile*, November, 1916.
"Other German Machines," *L'Aérophile*, March, 1916.

PROPELLER AND MACHINE GUN SYNCHRONIZING MACHINES.

"Fokker Firing Mechanism," *L'Aérophile*, June, 1916.
"Twin-Engined Machines."
L'Aérophile, July, 1916 (abstract in AVIATION AND AERONAUTICAL ENGINEERING, Sept. 15, 1916).
"Some Problems in Airplane Construction" (S. A. E. *Bulletin*, December, 1916).
"Army Aeronautical Specification No. 1002."
"The Development of the Military Aeroplane," by F. W. Lanchester. *Engineering*, March 3, 1916.

Chapter III

Estimates of Weight Distribution

Difficulties of the Subject

Hardly any branch of practical airplane design offers such difficulties as the estimate of weights. A manufacturer who has built a number of machines and has kept careful weight schedules has valuable data in his possession, but is, as a rule, chary of making such data public. Even an experienced manufacturer, however, may be at a loss when building an entirely new type, particularly if the new type is of a very different size from that to which he has been accustomed.

Theoretical considerations apply only to a limited extent. Empirical formulas have been suggested by several authorities, but are only partly satisfactory. The authors' thanks are due to manufacturers and others for such data as they have permission to publish.

Weight Shedules for a Machine of the Unarmed Tractor Reconnaissance Type (Two Seater) More Than 2500 Pounds in Weight

Full weight data can be published for one of the five machines which have been examined in Part 2, Section 1—the Standard H-3. The schedule for this machine is very complete, and is almost exactly in the form specified by the Aviation Section of the Signal Corps.

TABLE I.

Standard H-3, Hall-Scott A-5, 135 h.p. Maximum speed 84 m.p.h. Minimum speed 46 m.p.h. Climb 3400 ft. Total wing area, 532 sq. ft. Weight loaded, 2651.9 lb. (6 hours). Weight bare, 1908 lb.

Body structure:

Details:

Longeron, forward upper right	9.0 lb.
Longeron, rear upper right	9.0 "
Longeron, forward upper left	9.0 "
Longeron, rear upper left	9.0 "
Longeron, forward lower right	9.0 "
Longeron, rear lower right	9.0 "
Longeron, forward lower left	9.0 "
Longeron, rear lower left	9.0 "
Vertical posts, total	19.0 "
Rudder post	12.8 "
Horizontal posts, total	13.5 "
Engine beds (two)	19.0 "
Engine bed, supporting posts	4.0 "
Engine plates, total	19.0 "
Radiator supports	27.5 "
Fittings, total	27.5 "
Rivets, bolts, nuts, screws, total	7.1 "
Wire and cable, total with terminal clips and thimbles, etc.	21.1 "
Turnbuckles	10.0 "
Floor of cockpits	18.0 "
Tail skids	6.0 "
Body cover strips	7.0 "
Front and rear seats and supports	31.5 "
Cowling and body cover	59.5 "
Total	302.0 lb.

Percentage of total weight, 11.4%.

Chassis:

Details:

Wheels and tires, 2 at 26" x 5"	57.0 lb.
Axles (1)	22.0 "
Struts (2)	25.0 "
Axle braces	8.5 "
Axle mounting and guides	12.0 "
Rubber shock absorber	2.5 "
Fittings	7.5 "
Wiring and turnbuckles	2.5 "
Fairing	1.5 "
Total	138.5 lb.

Percentage of total weight, 5.23%.

Details for upper wings:

Front spar, total span 18' 5⅝"; 18' 8¾" length	25.5 lb.
Rear spar, total span 18' 9½"; 19' 0" length	23.5 "
Compression posts or solid ribs (7)	16.2 "
Lightened ribs and straps (10 long, 7 short), total	18.4 "
Entering edge, 4 pieces	8.0 "
Trailing edge, 4 pieces	3.9 "
Edge pieces and cross battens	5.5 "
Fittings	1.3 "
Wire, clips and turnbuckles	11.5 "
Linen, undoped, and tape and tacks for taping	15.4 "
Dope and varnish	6.5 "
Flaps uncovered, total	20.0 "
Flap hinges and hinge fittings, complete	1.1 "
Flap covering, dope and varnish	1.2 "
Unaccounted for	20.0 "
Total	178.0 lb.

Body wing section .... 16.5 lb.

Details for lower wings:

Front spar, total span 18' 5⅝"; 18' 8¾" length	25.0 lb.
Rear spar, total span 18' 9½"; 19' 0" length	23.0 "
Compression posts or solid ribs (8)	18.5 "
Lightened ribs and straps (7 short, 9 long)	19.5 "
Entering edge, 4 pieces	8.0 "
Trailing edge, 4 pieces	3.9 "
Edge pieces and cross battens, hinge, block braces, etc.	5.5 "
Fittings	1.3 "
Wire, clips and turnbuckles	11.8 "
Linen, undoped, and tape and tacks for taping	15.9 "
Dope and varnish	6.3 "
Flaps uncovored, total	20.0 "
Flap hinges and hinge fittngs, complete	1.1 "
Flap yokes and yoke fittings	3.8 "
Flap coverings, dope and varnish	1.2 "
Unaccounted for	25.3 "
Total	190.0 lb.

	Weight	Area	Weight per sq. ft.
Upper wings	178.0 lb.	262 sq. ft.	.680 lb.
Body section	16.5 "	18 " "	.920 "
Lower wings	190.0 "	262 " "	.720 "
Total wing weight	384.5 lb.		

Percentage of total weight, 14.52%.

Interplane struts, fittings, and wiring:	107.6 lb.
Weight per square foot of wing area	.203 "

Percentage of total weight, 4.06%.

Tail surfaces:

	Weight	Area	Weight per sq. ft.
Vertical fin complete, covered and varnished	3.0 lb.	5 sq. ft.	.600 lb.
Vertical rudder	9.0 "	10 " "	.900 "
Fixed horizontal tail	20.3 "	32 " "	.635 "
Elevators	21.0 "	23 " "	.915 "
Total	53.3 lb.		

Percentage of total, 2.0%.

Control system:

Combined Curtiss and Dep. control, with control operators in rear cockpit only	26.2 lb.
Control wires, wiring and switches	4.4 "
Total	30.6 lb.

Percentage of total weight, 1.15%.

Gasoline and oil:

Gasoline for 6 hours	396.00 lb.
Oil for 6 hours	33.5 "
Total	429.50 lb.

Percentage of total, 16.2%.

Tanks:

Tanks and connections and supports (68 gallons fuel) ... 78.5 lb.
Percentage of total, 2.95%.

Engine group:

Radiator, complete and connections without water..	46.0 lb.	Weight of radiator per hp..........	.34 lb.
Engine, complete without propeller, radiator or any water, any oil, long exhaust tube or self-starter.	558.5 "	Engine weight per hp	4.15 "
Water for radiator piping and jackets (30 lb. carried in radiator alone)...	93.8 "	Water weight per hp	.695 "
Propeller complete and bolts	27.5 "	Propeller weight per hp	.202 "
Long exhaust pipe.........	13.0 "		
Total...............	738.8 lb.		

Percentage of total weight, 27.77%.

Passenger and equipment:

Pilot	165.0 lb.
Passenger	165.0 "
Total............	330.0 lb.

Percentage of total, 12.5%.

Equipment:

Instruments and instrument board, and accessories compiled for rear cockpit............	22.7 lb.
Same for front cockpit............	9.2 "
Side pockets, both sides............	3.0 "
Camshaft oiler	4.1 "
Speaking tube	2.8 "
Pyrene and brackets complete............	6.7 "
Oil pressure line and sight oil............	2.8 "
Tool kit and case complete............	7.3 "
Total............	68.6 lb.

Percentage of total weight, 2.20%.

Summary of Weight Distribution for Standard H-3

Group	Weight	Percentage of Useful Load
Body assembly and equipment.....	370.5 lb.	13.60%
Chassis	138.5 "	5.23%
Wing group	384.5 "	14.52%
Interplane bracing	107.6 "	4.06%
Tail surfaces	53.3 "	2.00%
Control system	30.6 "	1.15%
Gasoline and oil............	429.5 "	16.20%
Gasoline tanks and piping.........	78.5 "	2.95%
Engine group............	738.8 "	27.70%
Passengers and equipment........	330.0 "	12.50%
Totals............	2651.9 lb.	100.00%

The Standard figures are fairly representative for this type of machine.

Percentage Table for Machines About 2500 Pounds

In Table 2 are given figures compiled by Dr. J. C. Hunsaker for a number of typical machines. The percentage values seem to hold very closely for machines of the large tractor type.

TABLE 2

Useful load:	Per cent
Personnel and equipment............	13.1
Gasoline and oil, 6 hours............	19.8
Engine weight:	
Tanks and pipes............	3.3
Engine and accessories............	17.9
Radiator (empty)	2.2
Cooling water	1.7
Propeller and hub............	1.0
Structural weights:	
Body	8.2
Landing carriage	8.2
Directive surfaces	4.1
Wings	16.5
Wing bracing	4.0
Total useful load............	32.9
" engine weight	26.1
" structural weight	41.0
	100

Weight Distribution for a Typical School Machine

Figures can be published for the Curtiss JN-4. Data was given in Section 1 for the JN-4B, but the difference between the two types is very slight.

Curtiss JN-4. OX 90 hp engine. Maximum speed 75 mph. Minimum speed 43 mph. Climbing speed 3000 ft. in 10 min. Total wing area, including wing flaps, 387.0 sq. ft. Weight loaded, 1902.35 lb. Weight bare, 1281.5 lb. 4.4 hours fuel.

Part:	Weight
Body assembly	290.00 lb.
Tail skid with rubber elastic cord (3 ft.)............	2.75 "
Cushions	3.50 "
Total............	296.25 lb.

Percentage of total load, 15.50%.

Chassis:

Details:

Landing gear braces with fittings............	28.5 lb.
Axle	15.75 "
2 Aluminum bearings for shock absorbers and straps....	2.00 "
2 Rubber shock absorbers (elastic cord 37 ft.)..........	3.50 "
2 Wheels 26" x 3" tire and 1½" hub............	27.00 "
Total chassis group............	76.75 lb.

Percentage of total weight, 4.03%.

Wing group:

Part	Weight	Area	Weight per sq. ft. of surface
Upper wings without flaps or fittings	120.00 lb.	172.2 sq. ft.	.772 lb.
Upper center section without fittings	13.00 "		
Lower wings without fittings.	112.00 "	152.2 " "	.736 "
2 wing flaps with fittings....	24.50 "	40.8 " "	.600 "
Total for wing group...	269.50 lb.		

Percentage of total weight, 14.15%.

Wing bracing and fittings:

Details:

Part	Weight
Upper wing fittings, 8 strut fittings, 4 fittings to center section	10.50 lb.
Lower wing fittings, 8 strut fittings, 4 fittings to body...	9.50 "
2 Wing skids............	2.25 "
4 Outer section struts (length of staggered struts = 61").	13.50 "
4 Intermediate section struts............	17.50 "
4 Drift wires to nose from upper and lower planes......	3.00 "
Flying landing and outer strut wires (not including center section)	24.00 "
4 Aileron wires with fittings............	5.00 "
4 Short uprights for bracing overhang............	1.00 "
Center section struts............	4.50 "
Center section brace wire............	3.50 "
Total............	94.25 lb.

Percentage of total load, 4.95%.

Tail surfaces:

	Weight	Area	Weight per sq. ft. of area
Vertical tail fin and fittings.	14.00 lb.	2.5 sq. ft.	0.62 lb.
Rudder and fittings.........	10.00 "	10.2 " "	1.02 "
Fixed horizontal tail with hinge fittings only........	14.00 "	22.7 " "	0.62 "
2 Elevator flaps with wires and posts	14.50 "	17.5 " "	0.83 "
Total............	52.50 lb.		

Percentage of total, 2.76%.

Control system:

Steering post, rudder wheels with rudder wires and fittings	12.25 lb.
4 Elevator wires............	3.50 "
Total............	15.75 lb.

Percentage of total weight, .83%.

Gasoline and oil:

4.4 hours	252 lb.

Percentage of total load, 13.20%.

Tanks:

	Weight	Weight of tank per gallon
Gasoline tank with capacity of 37 gall.....	28.1 lb.	.76 lb.

Percentage of total weight, 1.53%.

Engine group:

Propeller and hub.......	34.75 lb.	Weight per hp....	.386 lb.
Engine and accessories (including 2 hot air stoves, 3.75 lb., top of engine plates and side plate, 20.15 lb.)......	363.75 "	Weight per hp....	4.050 "
Radiator	55.75 "	Weight per hp....	0.620 "
Water	39.00 "	Weight per hp....	0.430 "
1 Water pipe and fittings	3.00 "		
Total............	496.25 lb.		

Percentage of total weight, 26.10%.

Passengers:

Pilot	156 lb.
Passenger	165 "
Total............	321 lb

Percentage of total weight, 16.95%.

Summary of Weight Distribution for JN-4-B

	Weight	Percentage of total load
Body assembly and equipment.....	296.25 lb.	15.50%
Chassis	076.75 "	4.03 "
Wings	269.50 "	14.15 "
Interplane bracing............	094.25 "	04.95 "
Tail surfaces	052.50 "	02.76 "
Control system	015.75 "	00.83 "
Gasoline and oil............	252.00 "	13.20 "
Gasoline tank and piping..........	028.10 "	01.53 "
Engine group............	496.25 "	26.10 "
Passengers	321.00 "	16.95 "
Totals............	1902.35 lb.	100.00%

Empirical Formulae and Values for Weight Estimates

Some empirical formulae and values are given here. Such empirical formulas can never be entirely trustworthy, since so much depends on the type of machine to be constructed, the type of construction to be employed for any particular part of the machine, and the factor of safety desired. Much greater reliance is to be placed on direct comparisons from actual machines and on actual computations from drawings. Still, they may serve a useful purpose in the preliminary stages of design, when a rapid estimate is needed.

(1) *Body*

Bare rectangular wooden longeron body, with fabric covering for small monoplane and biplane scouts about 1200 lb. total weight, 70 lb. is a good average figure. For large biplanes about 2500 lb. total weight, 150 lb. is a liberal allowance.

(2) *Seating*

About 10-12 lb. per person is sufficient.

(3) *Single control system 30 lb.*
Double control system 50 lb.

(4) *Landing Gear*

A landing gear of about $^1/_{16}$ the loaded weight of the machine can be easily designed.

(5) *Tail skid*

Is roughly $^1/_{20}$th the weight of the landing gear.

(6) *Main plane weights (surface alone)*

A fair average figure is 0.75 lb. per square foot of wing area, although wing weights will vary with size of machine, section employed, aspect ratio, strut spacing and numerous other features in design.

(7) *Weight of control surface*

Control surfaces with fittings and hinges may, with careful design, not exceed 0.5 or 0.6 lb. per square foot.

(8) *Tanks*

About 0.75 lb. to 1.00 lb. per gallon.

(9) *Engine weights, fuel consumption*

Full values for these are available and will be given in a subsequent section.

(10) *Engine mounting*

One-eighth of the engine weight for a rotary type and one-twelfth the engine weight for a fixed type engine.

(11) *Propellers*

A good rule is weight $= 2.5\sqrt{\text{hp}}$.

(12) *Radiators*

For radiators, manufacturers' figures will be given later, and empirical formulas are not necessary.

(13) *Passengers*

Some 10 lb. should be added for aviation dress.

(14) *Miscellaneous*

An allowance of 10 lb. is sufficient for instruments, such as compass, altimeters, etc. Fire extinguisher, 8 lb. Tool kit, 5 lb. First-aid kit, 5 lb.

(15)

In a subsequent section we shall deal with weights of such parts of the machine as cables, wires, turn-buckles, fabrics, dopes, wheels, etc., etc.

Some General Considerations on Distribution of Weight and Useful Load

F. W. Lanchester has approached the question of weight distribution for various sizes of machines in a very interesting article. The subject offers many difficulties, and the following notes, mainly based on Mr. Lanchester's article, are merely an introduction.

When estimating the structural weight of a new machine from data available on one constructed, certain theoretical considerations are available.

Simple and reasonable assumptions in dealing with the main planes are that the wing section remains geometrically similar, and the velocity constant. On such a basis from ordinary considerations of aerodynamics, the span must vary as the square root of the gross weight, and conversely the loading on the wing will vary as the square of the span. The direct forces of tension and compression on the spars will vary directly as the loading and square of the span, but the cross-sectional

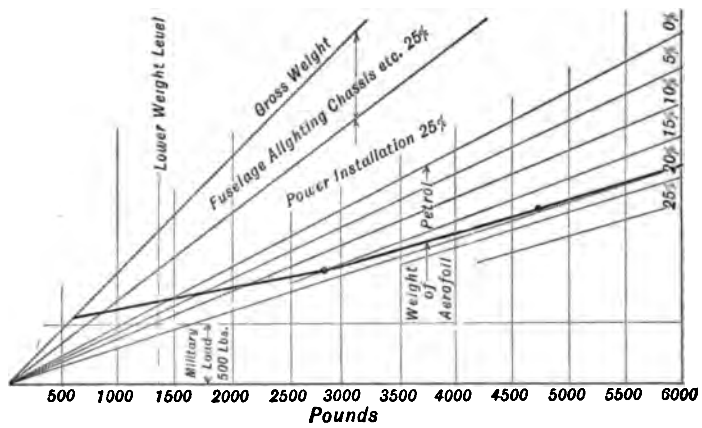

FIG. 1.

areas of the spars will also vary as the square of the span; geometrically similar wings will, therefore, be equally strong as regards direct forces. The bending moments will vary as the gross weight or loading multiplied by the span, i.e. as the cube of the span. But the resisting moment of the spars will vary as the section modulus or cube of the linear dimensions; geometrically similar wings will, therefore, be equally strong as regards bending moments. It follows that with constant velocities geometrically similar wings will be equally strong for both direct and bending stresses. From the weight of aerofoil point of view, the position is unfavorable, since the weight will vary as the cube of the linear dimensions or cube of the span.

It follows that the weight of the wings will vary as $W^{3/2}$ where W is the gross weight.

In the interplane bracing, the wires, which only take direct stresses, will be equally strong when geometric similarity is maintained. For the struts just as for the spars, the same will apply. Therefore, the interplane bracing will also vary as $W^{3/2}$.

For the body it is possible to make the somewhat more favorable assumption that its weight is directly proportional to the gross weight. With increase in span, it is by no means necessary to increase the arm of the tail surfaces proportionately, while the resisting moment of a body cross-section varies as the depth squared and the breadth. Current practice also seems to bear out the above assumption. It might even prove to be true that on large machines a slight saving on weight of body would be possible.

The shock of landing to be taken up by the chassis depends, for the same landing speed, on the gross weight.

If a chassis for a large machine were geometrically similar to that of a smaller machine, it would probably show greater strength in the struts, and equal strength in the shock absorber. The question is very complex. Mr. Lanchester insists on the analogy of the greater relative diameter of the legs of such a large animal as an elephant as compared with the legs of a flamingo. But with very big landing gear so much becomes possible in the way of shock absorption that keeping the weight of the chassis a constant proportion of the gross weight seems feasible.

For the power installation, no general discussion seems

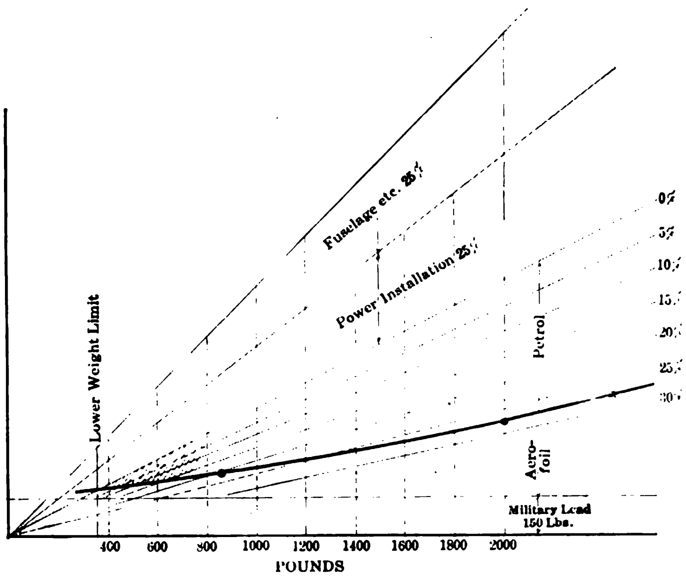

FIG. 2.

possible, and Mr. Lanchester has assumed this to be 25 per cent of the gross weight in the graphs of Fig. 1 and Fig. 2.

The construction of these is easily followed. The bounding line of these curves is drawn at 45° to the Case line, so that its ordinates represent the gross weight, just as the abscissae for the same point represent the gross weight. In accordance with the above considerations, the total structural weight is taken as a constant proportion of the gross weight, namely, 25 per cent. The power installation weight is taken as 25 per cent, as previously mentioned. The weight of aerofoil curve, varying as $W^{3/2}$, is obtained from present-day English practice in biplane construction, with a factor of safety of 6—somewhat lower than American practice. The military load is kept at 500 lb. in one case, at 150 lb. in the other, and the remainder is allotted to the supply of petrol.

The above remarks, the distribution of weights, and these two curves are open to criticism. However, they are the conclusions of a most eminent authority, and may serve as a useful guide in the preliminary design of a machine, particularly as regards possible endurance, which can at once be derived from the petrol capacity. They also give an idea of the limitations of the airplane. Thus the curves of Fig. 1 show the lowest possible weight level, with a big load of 500 lb., and if extended to greater gross weights would show where, with increased size, the petrol capacity begins to diminish. Fig. 2 would be particularly useful in considering the possibilities of a speed scout with a single passenger.

References for Part II, Chapter 3

"The Development of the Military Aeroplane," by F. W. Lanchester. *Engineering*, March 3, 1916.

Chapter IV
Engine and Radiator Data

General Requirements of Aeronautical Engines

The main requirements of an airplane engine are light weight, low fuel and oil consumption, reliability, accessibility, and a form suitable for installation in an airplane. The general form, apart from its weight, is important because of the question of mounting in the body, and the problem of engine cooling and body stream-lining. Selecting an engine for an airplane means unfortunately buying the engine most nearly suitable which is purchasable at the moment, and the choice is none too great. Nevertheless it is part of a designer's training to consider the comparative merits of every engine available, mainly with reference to the above points.

As regards reliability, no rules can be laid down. Satisfactory tests in Government or college laboratories are good guides. The reputation which an engine has earned among pilots under the more trying conditions of actual flying is even more important. Accessibility depends not only on the design of the engine itself, but on its careful mounting in the body. Fuel and oil consumption, weight and suitability of form are best studied by the compilation of such a table as Table 1. Such a table will require constant revision.

In considering weights of two engines of like power but of different type, such as a stationary air-cooled and a water-cooled engine, or a rotary air-cooled engine and a stationary water-cooled engine, radiator and cooling water should not be neglected. In dealing with rotary engines, fuel and oil consumption tend to make comparisons with stationary engines less favorable to the former type than is at first apparent. Particularly is this the case when a flight of more than 2½ or 3 hours duration is contemplated. The extra weight of gasoline and oil to be carried for the rotary may actually make it the heavier engine at the beginning of a fairly long flight.

The form of an engine, from the points of view of mounting and projected area, are best studied from drawings appearing in technical magazines and makers' catalogs. The dimensions given in the table serve as a preliminary guide in narrowing down selection.

For a general study of the subject of aeronautical engines reference is appended to one or two excellent books—in which, however, no information as to recent developments is available.

The question of revolutions per minute apart from the question of power and efficiency in the engine itself has an important bearing on propeller design. Wooden propellers of large diameter seem to reach their maximum permissible safe speed at about 1300 r.p.m. Beyond this figure, it is hard to keep stresses down. Questions of direct drive and geared-down drives must be considered from this point of view.

Acknowledgment is made to Lieutenant H. C. Child, and to Mr. Lee S. Wallace for valuable data.

Weights for Radiators and Cooling Water

The following are good preliminary figures for design in accordance with general data:

Bare radiator	.55 lb. per bhp.
Water in radiator	.13 lb. per bhp.

The Ajax radiator employed in conjunction with a 130-hp. Hall-Scott engine weighs 45 lb. bare and carries 30 lb. of water. On a school Curtiss of the JN type with a 90 hp. Curtiss engine, the figures for a Rome-Turney radiator are

Weight of empty radiator	58½ lb.
Weight of water contents	24¾ lb.
Thickness of core	2⅞ in.
Active front area	400 sq. in.
Total radiating surface	15,360 sq. in.

For the Livingston Radiator, the following information is available:

"For a 120 h.p. engine, from 16,000 to 18,000 sq. in. of radiator surface is required. Each square inch of projected area of 4 in. section contains 50 sq. in. of cooling surface. A 5 in. section contains 60 sq. in., and a 3 in. section contains 40 sq. in. Therefore, a radiator for a 125 h.p. engine will have between 320 sq. in. (2.2 sq. ft.) and 360 sq. in. (2.5 sq. ft.) projected area of 4 in. section. A radiator for such an engine contains approximately 4 gallons of water. Of this, 1 gallon is contained in the cells, the other 3 gallons in the headers. The headers should be of such proportions that the lower has about two-thirds the capacity of the upper."

Practical Rules for Cooling Surface for Radiator of Honeycomb Type

From Dr. Hunsaker's experiments at the Massachusetts Institute of Technology, and certain theoretical considerations, a surface of .83 sq. ft. per bhp. has been found necessary for the honeycomb type. C. Sage recommends 1.08 sq. ft. per bhp. for an airplane of an average speed of 60 m.p.h., and presumably a minimum speed of 45 m.p.h. An allowance of 1 sq. ft. per bhp. seems very fair for machines of medium speed. In fast machines of the pursuit type, it would be possible to go considerably below this figure; even if a fast machine makes a prolonged climb, it will never do so at its slowest speed. Dr. Hunsaker has shown that an empirical formula may be established of this type:

$$a = \frac{C \times bhp.}{V}$$

where a = area of cooling surface, C is some constant and V is the speed in miles per hour. A designer, who has satis-

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Maker and Model	No. Cylinders.	Type.	Rated H.P.	R.P.M.	R.P.M. of Propeller.	Gas per H.P.-hr. in lb.	Oil per H.P.-hr. in lb.	Weight Empty with Ignition and Carburetor, lb.	Weight Cooling Water in Radiator and Engine, lb.	Weight of Example of Radiator, lb.	Weight Cooling Water in Radiator, lb.	Weight of Example of Muffler, lb.	Weight Self-starter, lb.
Aeromarine	6	Vert.	85	1400	1150	.7	.069	440	37.5	31	..	8.12	..
Aeromarine D-12	12	V.	160	1400	..	..	..	750	..	..	..	..	..
Aeromarine	8	V.	100	..	..	..	..	410	..	..	..	..	..
Atwood C-12	12	V.	150	2500	1250	.64	..	592	..	63	..	13.25	..
Christofferson	6	Vert.	113	1450	1450	.665	.0266	510	..	67.5	..	8.7	40
"	8	V.	160	2500	..	..	..	630	65	85	..	15	17.5
Curtiss-Ox 2	8	V.	90	1400	1400	..	..	375	..	58¼	24¾	..	..
" -OXX 2	8	V.	100	1400	1400	.59	.035	423	44	76	..	12.25	46
" -VX	8	V.	160	1400	1400	.575	.0328	645	62	86	..	16	72
" -VX 3	8	V.	200	..	..	.73	.031	667	70	94	..	16	55
" -V-4	12	V.	250	..	..	.612	..	1125	100	120	..	24	..
Duesenberg	4	Vert.	140	2100	..	..	..	455	..	..	..	7.21	..
"	12	V.	250	1800	..	..	..	425	..	..	..	15.5	..
Genl. Ordnance Co.	8	V.	250	1800	..	..	..	867	..	..	..	..	..
Genl. Vehicle Gnome	9	Roty.	100	1200	..	.72	..	272	..	..	..	..	..
Gyro K	7	Roty.	90	1250	..	.545	.166	242	..	..	..	..	..
" L	9	Roty.	100	1200	..	.614	.180	285	..	..	..	..	..
Hall-Scott A-7	4	Vert.	80-90	1370	1370	.47	.037	410	34*	40	..	..	..
" A-5	6	Vert.	125	1300	1300	.507	.028	592	52*	45	30	..	42
" A-5a†	6	Vert.	162	1325	..	14.7‡	.4‡	562	..	..	..	..	..
Hispano-Suiza	8	V.	154	1500	..	..	..	455	..	48.3	..	10	..
Knox	12	V.	300	1800	..	..	..	1425	..	..	..	19.4	78
Packard	12	V.	225	2100	..	..	..	800	..	..	..	..	..
Rausenberger C-12	12	V.	150	1300	..	..	..	570	..	..	..	..	..
Sturtevant 5	8	V.	140	2000	..	.54	.043	600	70	60	40	..	50
" 5-A	8	V.	140	2000	1200	..	..	514	66	54	40	..	..
"	12	V.	275	2000	..	..	..	..	..	..	..	..	90
Thomas 8	8	V.	135	2000	1200	.59	.053	572	80	100	..	15.3	50
" 88	8	V.	150	2000	..	.59	..	485	..	..	..	..	..
Wisconsin	6	Vert.	140	1380	1380	.550	.027	637	38*	50	25	..	..
"	12	V.	250	1200	1200	.602	.029	1000	..	142	..	18.1	47
Wright	6	Vert.	60	1400	..	.545	..	306	..	39	..	6.25	..

**Engine only.*
† Figures obtained from test run.
‡ Gallons per hour.

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tion

Propeller, lb.	Weight of Example of Mounting, lb.	Weight of Example of Cooling, lb.	Bore in inches.	Stroke in inches.	Piston Displacement, cu. in.	Overall Length in in.	Overall Width in in.	Overall Height in in.	No Supporting Brackets.	Fore and aft distance between Brackets.	Athwartship distance between Brackets.	Athwartship distance between Bolt Holes.	Maker and Model
.8	28.2	5.5	4 5/16	5⅝	449	58	30	36¾	..	..	..	..	Aeromarine
.	..	..	..	..	..	67	24	30	3	..	..	..	Aeromarine D-12
.	..	..	..	..	..	..	..	..	..	..	..	..	Aeromarine
8	55	40.9	3½	4½	519.5	48	27	26	..	..	..	..	Atwood C-12
.5	34	6.5	4¾	6	..	..	..	..	..	..	..	..	Christofferson
0	41	17.5	..	..	..	..	..	..	..	..	..	..	"
.	..	..	4	5	502	50	30	27	3	9	..	12⅞	Curtiss-Ox 2
.5	23.2	10.6	4¼	5	567.5	50	30	29	3	9	..	12⅞	" -OXX 2
4	45	12.2	5	7	1100	67⅜	35	34	5	9	14	16	" -VX
4	14.7	12.2	5	7	..	..	..	..	..	..	..	..	" -VX 3
5	60.1	19.6	5	7	..	84⅝	34⅛	40	..	..	..	..	" -V-4
.	29.5	4.8	4⅞	7	496	43⅜	15½	37⅜	4	13⅝	13⅝	14¼	Duesenberg
.	49.2	18	4⅞	7	1488	68	31¼	39⅝	4	15⅜	14	15½	"
.	..	..	4¾	6½	920	..	..	..	..	..	..	..	Genl. Ordnance Co.
.	..	..	4.33	6½	920	..	..	..	..	..	..	..	Genl. Vehicle Gnome
.2	22.7	12.4	4½	6	..	..	..	..	..	..	..	..	Gyro K
.6	24.4	13.2	4½	6	859	24.7	37½	37½	..	..	..	..	" L
7	28.9	4.5	5	7	550	57	18½	39½	4	16¼	14	16	Hall-Scott A-7
.5	39.6	6.8	5	7	825	63⅜	18½	41¼	3	35¼	14	16	" A-5
.	..	..	..	..	..	..	..	..	..	..	..	..	" A-5a†
7	13.5	28.3	4⅝	5	672	52½	32½	35½	12¼	14⅜	..	..	Hispano-Suiza
.	56.8	21	4¾	7	1555	..	..	..	..	..	..	..	Knox
.	..	..	4	6	905	62¾	23 1/16	41½	3	..	16¼	18¼	Packard
.	..	..	4⅛	6	900	42	24	..	..	..	..	..	Rausenbergér C-12
0	30	30	4	5½	522	59 5/16	23	29 9/16	3	14 1/16	..	13¾	Sturtevant 5
.	..	..	4	5½	522	59 5/16	34	34 9/16	3	14 1/16	..	..	" 5-A
4	..	..	4	5½	..	..	..	..	..	..	..	..	"
.8	40.1	13.6	4	5½	552.9	60	28	37⅞	3	13 1/16	..	13¾	Thomas 8
.	..	..	..	..	..	..	..	..	3	13 1/16	..	13¾	" 88
2	..	..	5	5½	765.7	17¼	24	41 5/16	4	14½	13¾	16½	Wisconsin
.	54.2	10.8	5	6½	1532	76 3/16	37⅛	37 1/16	4	14¾	15½	18½	"
9	25.9	6.4	4⅜	4½	..	..	..	..	..	..	..	..	Wright

factory data on a machine of a certain speed, can employ this rule for machines of a different speed.

Position and Resistance of a Radiator

Tests at the Massachusetts Institute of Technology show that the resistance of a radiator may be represented by the equation

$$R = K_x A V^2$$

where R = resistance in pounds

A = area of radiator face in square feet

V = speed in miles per hour.

and $K_x = .00175$

While these tests were conducted on very small sections, the results are safely applicable to full-size radiators.

Figures on the resistance of a given section of radiator in a current of air do not by any means settle the problem of the best position for the radiator. Manufacturers have placed radiators in various positions, claiming minimum resistance for each position. If a radiator is placed behind the propeller where the slip stream increases the velocity by some 25 per cent, the cooling surface may be decreased by 25 per cent with a consequent reduction of resistance producing area, but since the resistance varies as the square of the velocity, there is finally an increase of 25 per cent. These or similar considerations have led designers to place radiators underneath the wings. But it is forgotten that when a radiator is placed underneath the wings, it is no longer a shelter for the body. There is also the question of extra length of piping. For radiators placed at the sides, Dr. Hunsaker's opinion is that a more generous allowance is necessary. Dr. Zahm's skin friction formula is of the form $R = 0.0000158\, l^{.93}\, V^{1.86}\, b$. as we saw in Part 1, Section 3, AVIATION AND AERONAUTICAL as we saw in Part 1, Chapter 3, where l is the length of surface parallel to the wind. Owing to the greater length of side radiators in the direction of motion, they are therefore probably less effective.

The authors' opinion is that the best and most natural position of the radiator is behind the propeller, but the question is hardly capable of a decision so far.

Practical Construction of Radiators

C. Sage, engineer of the Rome-Turney Radiator Company, has submitted the following authoritative views:

"As to the construction of radiators, we may say that the simpler the outline the more durable will be the radiator—and the cheaper. The cooling section of honeycomb radiators ought to have outlines composed entirely of straight lines—curves in the honeycomb are expensive and are weak points for the reason that all honeycomb cooling sections are at the start made of rectangular blocks and then sawed to shape on a band saw like a board. The sawed-off waterways have then to be patched up again with solder and their ends are naturally not as strong as before. Then this section has to be fitted to the case and the more curves there are the more difficult and costly the fitting will be. Concerning the case of the radiator the same principle holds good—the simpler the design the better and cheaper the product. All curved surfaces are costly if they have to be produced by hand work and pressed cases must be made in large quantities in order to pay for the necessary punches and dies and it takes a long time until production can be started on them.

"As for water connections between engine to radiator and pump to radiator, it is very important that they be large enough to convey the large bulk of water with the least possible pressure. If the connections are too small a considerable vacuum will be set up in the suction line from the radiator to the pump and consequently air will be drawn into this line at all leaky points, prominent among these being the stuffing box and the grease cups of the pump. This air will be mixed with the water forming a milky liquid like charged water, increasing its volume, and consequently a considerable loss through the vent pipe will take place.

"As to support of a radiator, the most satisfactory method is the use of a cradle or cross piece at the front of the body, in the case of a tractor, on which the radiator is placed and fastened by studs in the bottom tank. In the case of pushers, many different suspension methods are used, none of which can be called standard, and the same is true for side radiators."

An excellent point made by the manufacturers of the Ajax radiator is the reinforcement of the fins rear and front by soldering on 1/16 in. wires. Dividing a radiator into two parts by a small ¾ in. deep water tank to permit settling of the water, is another good point in this type.

As a general rule, the sub-division of a radiator into a number of sections is advantageous. In the present ill-defined position of radiator design, it is an advantage to be able to increase or diminish the radiator surface of a given machine.

References for Part II, Chapter 4

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"Notes on Radiator Design," by J. C. Hunsaker, *Aerial Age*, May 29, 1916.

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Chapter V

Materials in Airplane Construction

Within the limits of one chapter it is impossible to treat adequately all the data on materials required for airplane construction. The data included here will be sufficient for the purpose of our design, however, and a number of references are appended. For practical work, the designer must procure all necessary handbooks, make tests of his own special fittings, and generally collect his own data.

Special Utility of Wood in Airplane Construction

It is the remarkable strength for its weight which makes wood so useful in airplane construction. If we compare spruce, weight per cubic foot 26 lb., tensile strength 9000 lb., with mild steel weighing 490 lb. per cubic foot with a tensile strength of 60,000 lb., the spruce will be $\frac{9000}{60,000} \times \frac{490}{26} = 2.9$ times as strong for the same weight.

The selection, mechanical properties and correct structural employment of timber are, however, inexhaustible subjects, and the following notes are the barest summary of the factors the designer must have in mind.

Weight of Wood

The weight of wood varies greatly for the same species and for portions of the same tree. Sapwood is heavier than heartwood, summerwood than springwood. Green timber naturally weighs more than dry timber, due to the presence of sap and moisture. The ultimate wood fiber of all species has a specific gravity of 1.6, so that no wood would float in water were it not for the buoyancy of the air present in the cells and walls.

TABLE 1

Specific Gravity and Weights of Woods

Dry woods	Wt. per cu. ft. lb.	Specific gravity
Ash, American white	38.	.610
Balsa	6.5	.104
Boxwood	60.	.960
Cherry	42.	.672
Chestnut	41.	.660
Cork	15.	.250
Elm	35.	.560
Ebony	76.1	1.220
Hemlock	25.	.400
Hickory	53.	.850
Lignum-vitæ	83.	1.330
Mahogany, Spanish	53.	.850
Mahogany, Honduras	35.	.560
Maple	49.	.790
Oak, live	59.3	.950
Oak, white	48.	.770
Oak, red	40.	.640
Pine, white	25.	.400
Pine, yellow	34.3	.550
Pine, southern	45.	.720
Sycamore	37.	.590
Spruce	25.	.400
Walnut	38.	.610

The weight of wood is experimentally determined by subjecting thin discs to an oven temperature of 100° Cent. until they cease to lose weight by evaporation of moisture. But even with this provision, the results will be extremely variable, and the value usually assigned to a given species is simply the average of a large number of tests. Table 1, taken from a Bulletin of the Forestry Division, United States Department of Agriculture, will give values sufficiently accurate for design.

Weight is a good indication of the strength of wood, provided the amount of moisture contained is known. As a general rule, we may say that a comparison of two woods, each containing the same percentage of moisture, will show the heavier to be the stronger; in fact, the strength will be very nearly proportional to the weight.

Factors in the Mechanical Properties of Woods

The strength properties of wood depend on (1) correct identification of species and variety; (2) age and rate of growth; (3) position of test specimens in the tree; (4) moisture content; (5) relative freedom from defects, such as knots, etc.

Tensile Strength

Tensile tests are difficult because tests cannot be devised that do not involve either shear along the grain or compression across the grain. It is for the same reasons that wood may be unsuitable in tension, though it is apparently strong under such a stress.

Failure in tension along the grain involves principally the resistance offered by the wood elements to being torn apart transversely or obliquely. The strands of wood elements are practically never pulled apart by failure of the union between adjacent strands or fibers.

Cross grain is prejudicial to tensile strength and rays, owing to their transverse position with respect to a load applied along the grain, and small resistance to tension in a direction normal to the direction of their fibers greatly weaken the timber. Knots weaken wood subjected to longitudinal tension.

Compressive Strength

Individual fibers act as so many hollow columns bound firmly together, and failure involves either buckling or bending of the individual fibers or bundles of elements which finally come to act almost independently.

Compressive strength depends on a number of factors: (1) density; (2) strength of union between individual fibers as affected by moisture content; (3) stiffness of wood fibers (again largely a matter of moisture content); (4) continuity of the course of longitudinal strands in a direction parallel to axis of the piece. Woods in which separate elements are closely interlaced and bound together will be stronger than woods of opposite character.

The strongest woods in compression with the grain are, roughly, in the following classes:

(1) The dense and tough hickory, birch, hard maple, etc.; (2) oak, elm, ash; (3) spruce, pine and fir.

Crushing Across the Grain

Crushing strength across the grain is dependent practically entirely upon the density of the wood. Crushing strength across the grain is, therefore, least for the lightest, most porous woods, and greatest for heaviest and densest woods.

Compressive strength across the grain is to compressive strength along the grain as 13 to 14 per cent for white pine, cedar, cypress and spruce, 15 to 16 per cent for the various grades of hard pine, 18 to 26 per cent for elms, 21 to 26 per cent for ash, 22 to 26 per cent for oaks, 23 to 31 per cent for hickories.

Strength in Bending

In considering the strength of a wooden beam in bending, several difficulties occur. Longitudinal shear is very important. A wing spar may be amply strong in bending, and yet if highly channeled out fail by longitudinal shear. The tensile fiber strength of wood is much in excess of the compressive stress, but even if the compressive fiber stress of wood is employed in the formula $f = \frac{M_y}{I}$, it is no true criterion. If this formula is employed for strength computations in bending, it is assumed that the material is still behaving elastically up to actual failure, and therefore that the fiber stress is still directly proportional to the distance of the fiber from the neutral axis. As a matter of fact, the elastic limit of the material may have long been passed when the breaking load is reached, the neutral axis may have shifted, and the extreme fiber may be no longer proportional to the bending. Therefore, in stress calculations for wing spars, these considerations should be applied, making use of the modulus of rupture—for which values are given in Table 2—deduced from actual bending tests, which are far more trustworthy guides.

Knots

Knots originate in the timber cut from the stem or branches of a tree because of the encasement of a limb, either living or dead, by the successive animal layers of wood. Most limbs originate at the pith of the stem, and the knots found deep in a log are therefore small, increasing in size toward the bark. So long as the limb is growing, its layers of wood are a continuation of those of the stem. But a majority of the limbs die after a time, and if a portion of a dead limb is subsequently encased by the growing stem, there will be no intimate connection between the new stem wood and the dead wood of the limb, and a board so cut as to intercept this portion of the log will contain a loose knot. A board cut from the log at such a depth that the limb is intercepted at a point where it was encased while still living will contain a sound knot, unless the knot has rotted, become badly checked, or contains a large pith cavity.

A sound knot is usually harder than the surrounding wood, and in coniferous woods is apt to be very resinous. On this account it may constitute a defect because of its non-retentivity of paint or varnish. Otherwise it constitutes a defect only on account of the disturbance to the grain and difficulty caused in working, or in the event of its occurrence on the under side of a timber used as a beam, a weak point exists, owing to its small resistance to tensile stress. A knot constitutes an impediment to the splitting of timber, since the fibers of the stem wood above a limb bend aside and pass around the limb, while the fibers below run continuously into the limb. Thus it often happens that a cleft started above a limb will never run into a knot, but one started below is very apt to do so.

The Effect of Moisture on Strength of Wood

Loss of moisture does not affect the strength of wood in any way, until the total moisture content has been reduced below the critical percentage, which represents the fiber-saturation period. Beyond this point, progressive loss of moisture affects the strength very considerably. Thus the strength of green wood is only 50 to 60 per cent of normal air-dry conditions (12 per cent moisture), while the strength of kiln-dry wood exceeds the strength of air-dry woods by some 50 to 70 per cent.

Time Factor in Tests of Timber

Timber differs from most other materials in that small variations in the rate of application of load have a more pronounced effect upon the strength and stiffness shown by a specimen under test. If a timber-compression block or beam is loaded rapidly, it will appear to have a higher elastic limit and ultimate strength, and will also appear to be stiffer, than it will if loaded less rapidly. This is due to the fact that the deformation lags far behind the load, and if any load is permitted to remain upon a specimen for a time the deformation increases, the amount of increase becoming greater for heavier loads. Actual failure appears to be consequent upon the attainment of a certain limiting amount of deformation or strain, rather than a limiting load or stress.

Difficulties of Wood Construction in Airplanes

The comparative values of Table 2 demand the most careful study. A certain type of timber may be most suitable for the direct stress to which it is subjected, yet fail completely under certain indirect stresses, either inherent in the construction or due to faulty design. For example, at the hinging of a wing spar to the body, if the bolts are not correctly placed, they may shear out the wood. These points will be considered in detail in the design, but enough has been said to show the value of studying not only the direct stresses on a piece of timber in a machine, but also the indirect stresses producing crushing across the grain, shear, etc.

Strength Values for Timber

In no material are such conflicting values given by various authorities as for timber. The size of the specimen under test, the dryness, the method of applying the load, and its previous history, all tend to introduce discrepancies. Until the Bureau of Standards, or some other testing laboratory, has gone thoroughly into the question, all the values employed by airplane constructors will be open to suspicion. Table 2 is a summary of information taken from various sources. This table is not unimpeachable, but it approximates closely values used in current practice. In airplane design, fiber stress is still taken as a criterion, without due consideration of the modulus of rupture.

Acknowledgement is due Prof. W. H. Keith for collaboration on brief notes on timber.

TABLE 2

Material.	Tensile strength with grain.	Tensile strength across grain.	Compressive strength with grain.	Compressive strength perpendicular to grain (fiber stress at elastic limit only).	Shearing strength parallel to grain.	Hardness load required to embed a 0.444 in. ball to half its diameter.	Modulus of rupture in static bending.
Ash.......	12,000	1,600	9,000	1,300	1,500	56.5	15,660
Balsa......	11,000	1,600	1,200	..	..	60–70	..
Beech.....	11,000	1,600	8,600	..	1,200	..	10,000
Birch......	15,000	1,800	3,500	..	1,050	..	11,700
Cedar.....	10,800	700	5,700	..	..	..	7,400
Elm.......	10,000	..	7,000	..	..	..	8,800
Hickory...	15,000	2,200	11,000	944	1,500	..	15,000
Mahogany.	16,000	..	8,200	..	..	..	11,000
Maple.....	11,150	1,500	7,150	606	1,130	..	12,000
Oak.......	15,000	2,000	7,000	..	1,215	1,139	10,600
Pine, yellow	13,000	1,100	5,400	342	640	310	4,760
Pine, white	10,000	1,100	5,000	314	640	304	5,000
Spruce.....	9,000	600	6,500	400	500	272	9,200

Wires and Cables

The following terms are in common use: (1) "Solid wire stay" or "aviation wire" of one wire of suitable diameter; (2) "strand stay," consisting of either 7 or 14 wires stranded together and known to the trade as "aviation strand"; (3) "cord" or "rope stay," consisting of 7 strands twisted together, forming a rope, the strands being either 7 wires or 19 wires; (4) "flexible cord," composed of six strands of seven wires, with a center of either cotton or wire, as ordered. The cord with the cotton center is considerably more pliable than that with the wire center.

Vanadium steels and other special steels have not as yet become established as desirable wire steels, and carefully made high-grade carbon steel is at present most largely employed in the manufacture of wires and cables.

Properties of Metals

Only the briefest outline can be given of the metals that are commonly employed in airplane construction. To enter into any adequate discussion of this branch of the work would require a book in itself. The constructor must keep constantly before him some standard book on this subject and at the same time resort to strength of material and part tests whenever new combinations are to be employed in his design.

The following table of weight and melting points for various metals may be of service:

Metal.	Weight per cu. in.	Weight per cu. ft.	Specific Gravity.	Melting point Fahr. °.
Aluminum............	.096	166	2.66	1,215
Antimony............	.242	418	6.70	786
Bismuth.............	.350	607	9.74	516
Brass, cast...........	.292	504	8.10	1,635
Brass, rolled..........	.303	524	8.40	....
Bronze, gunmetal.....	.305	529	8.50	1,800
Copper, cast..........	.314	542	8.70	1,980
Copper, cold.........	.321	555	8.90	....
Duralumin...........	.103	178	2.85	(approx.)1,300
Gold, 24 carat........	.694	1,204	19.26	1,950
Iron, cast............	.260	450	7.21	2,012
Iron, wrought........	.278	480	7.70	2,912
Lead, cast............	.410	710	11.38	621
Mercury 60° Fahr.....	.489	846	13.58	....
Monel metal.........	.320	553	8.85	2,480
Platinum.............	.779	1,342	21.50	3,236
Silver................	.378	655	10.50	1,762
Steel, rolled..........	.283	490	7.85	2,552
Tin, cast.............	.266	459	7.35	450
Zinc, cast............	.248	429	6.88	786

Strength and Weights for Wire and Cables

TABLE 3

ROEBLING SOLID WIRE

Diameter of cord, Inches	Breaking strength of cord, Pounds	Approximate weight per 100 ft., Pounds
1/16	400	.73
5/69	480	.83
3/32	780	1.30
7/64	830	1.50
1/8	1,150	2.20
5/32	2,200	4.20
3/16	2,750	5.30
7/32	4,000	7.43
1/4	5,000	4.50
5/16	7,900	15.00

ROEBLING'S 19 WIRE GALVANIZED—AVIATOR WIRE STRAND

1/32 (7 wire)	185	0.30
1/16	500	0.78
5/64	780	1.21
3/32	1,100	1.75
7/64	1,600	2.60
1/18	2,000	2.88
5/32	2,800	4.44
3/16	4,200	6.47
7/32	5,600	9.50
1/4	7,000	12.00
9/32	8,000	14.56
5/16	9,800	17.71
11/32	12,500	22.53
3/8	14,400	26.45

AMERICAN STEEL AND WIRE CO. GALVANIZED (19 WIRE) AIRPLANE STRAND

1/32 (7 wires)	125	.23
1/16	500	.89
3/32	1,100	1.70
1/8	2,000	3.3
5/32	3,000	5.1

ROEBLING'S 7 BY 19 HEAVILY TINNED—AVIATOR CORD

1/8	2,000	2.88
5/32	2,800	4.44
3/16	4,200	6.47
7/32	5,600	9.50
1/4	7,000	12.00
9/32	8,000	14.56
5/16	9,800	17.71
11/32	12,500	22.53
3/8	14,400	26.45

ROEBLING EXTRA FLEXIBLE AVIATOR CORD, 6 x 7 COTTON CENTER

5/16	9,200	16.70
1/4	5,800	10.50
7/32	4,600	8.30
3/16	3,200	5.80
5/32	2,600	4.67
1/8	1,350	2.45
7/64	970	1.75
3/32	920	1.45
5.64	550	.93
1/16	485	.81

AMERICAN STEEL AND WIRE CO. GALVANIZED OR TURNED FLEXIBLE CORD

3/16 (19 x 7)	2,600	5.52
5/12 (19 x 7)	1,800	3.85
1/8 (19 x 3)	1,150	2.45
3/32 (12 x 3)	725	1.55

The figures given above have been revised, and the following table will supplement the previous values.

ROEBLING TINNED AIRCRAFT WIRE

American gauge (B&S) Number	Diameter, in.	Minimum breaking strength, lb.	Weight lb. per 100 ft.
8	.128	3000	4.40
9	.114	2500	3.50
10	.102	2000	2.77
11	.091	1620	2.20
12	.081	1300	1.744
13	.072	1040	1.383
14	.064	830	1.097
15	.057	660	.870
16	.051	540	.690
17	.045	425	.547
18	.040	340	.434
19	.036	280	.344
20	.032	225	.273
21	.028	175	.216

ROEBLING 19-WIRE GALVANIZED AIRCRAFT STRAND

Diameter of strand, in.	Breaking strength of strand, lb.	Approximate weight, lb. per 100 ft
5/16	12,500	20.65
1/4	8,000	13.50
7/32	6,100	10.00
3/16	4,600	7.70
5/32	3,200	5.50
1/8	2,100	3.50
7/64	1,600	2.60
3/32	1,100	1.75
5/64	780	1.21
1/16	500.	.78
1/32 7 wire	185	.30

Roebling 6 x 7 (Cotton Center) Galvanized Aircraft Cord

Diameter of cord, in.	Breaking strength of cord, lb.	Approximate weight, lb. per 100 ft.
5/16	7,900	15.00
1/4	5,000	9.50
7/32	4,000	7.43
3/16	2,750	5.30
5/32	2,200	4.20
1/8	1,150	2.20
7/64	830	1.50
3/32	780	1.30
5/64	480	.83
1/16	400	.73

Roebling 7 x 7 (Wire Center) Galvanized Aircraft Cord

Diameter of cord, in.	Breaking strength of cord, lb.	Approximate weight, lb. per 100 ft
5/16	9,200	16 70
1/4	5,800	10.50
7/32	4,600	8.30
3/16	3,200	5.80
5/32	2,600	4.67
1/8	1,350	2.45
7/64	970	1.75
3/32	920	1.45
5/64	550	.93
1/16	485	.81

Roebling 7 x 19 Tinned Aircraft Cord

Diameter of cord, in.	Breaking strength of cord, lb.	Approximate weight, lb. per 100 ft.
3/8	14,400	26.45
11/32	12,500	22.53
5/16	9,800	17.71
9/32	8,000	14.56
1/4	7,000	12.00
7/32	5,600	9.50
3/16	4,200	6.47
5/32	2,800	4.44
1/8	2,000	2.88

Wire, Strand or Cord

Roebling's report does not settle the question as to which is a correct selection. A comparative table shows the progressive decrease in strength.

Material.	Diameter.	Strength of material.	Strength of stay
Wire	3/16	5,500	5,100
Strand	3/16	4,600	4,100
7 by 19 cord	3/16	4,200	3,500

A stranded or cord stay has about 20 per cent more aerodynamical resistance than a solid wire of about the same diameter. There appears to be a slight advantage in favor of solid wire as regards strength of stay. On the other hand, the strand stay is one and a third times more elastic, the cord one and three-quarter times more elastic than the solid wire. In American practice all three types of stays appear. No doubt the use of strand or cord is justified by the extra elastic stretch and flexibility.

Exact data on the fatigue values of the three materials is not available, but it is well known that strand or cord will stand up much better to vibration than a wire. Also in a continuous beam structure such as that of a wing, there may be deflections of unknown magnitude, in which case the more elastic cable will be somewhat safer. On the other hand an exposed cable is liable to damage. A single small wire of one strand may be damaged and lead to the eventual destruction of the whole cable. In the covered in body a cable is not likely to be damaged. The problem is by no means settled yet, and only comparative experience in actual flight and further experimentation can give a definite solution.

Turnbuckles

The breaking strength of turnbuckles made of precisely the same material will vary enormously with every type of construction, and the makers' catalog or data sheet has to be consulted for every special case. In Figs. 1 and 2 are shown two representative types, the Curtiss and the National, Burgess, Meyer, Binet types. The weights and strength values are fairly representative of what can be obtained from this important brand of airplane material. In Table 4 are given results of tests on some specimens of the Standard Screw Co. of Pennsylvania:

TABLE 4

Standard Screw Co. Turnbuckles

Manufacturer number,	Mean breaking load.	Mean breaking load from test.	Initial shank area per sq. in.	Weight in lb Long,barrel male
326	2,150	2,432	.01864	.122
327	2,850	3,496	.0260	.167
328	3,500	4,605	.0350	.232
329	5,000	6,545	.0515	.275
330	840	9,530	.0794	.411

Mean tensile strength of shank, 128.610 lb. per sq. in. Material—shank, 3½ per cent. nickel steel, heat treated; barrel, robin bronze. Turnbuckles are made in short and long male and female ends.

Strength of Steel—Pounds per Sq. Inch

	Tension Ultimate	yield point.	Compression Ultimate	yield point.	Shearing Ultimate	yield point.
0.05% C for rivets.....	45,000	22,000	70,000	40,000	45,000	22,000
0.10% C boiler plate...	55,000	30.000	95,000	65,000	50,000	28,000
0.25% C structural....	65,000	34,000	110,000	85,000	55,000	32,000
0.40% C rails.........	70,000	45,000	120,000	90,000	60,000	35,000
0.90% C machinery...	90,000	70,000*	140,000	110,000*	70,000	*45,000
1.00 to 2.00% tool.....	150,000	none	200,000	none	120,000	none

*When well annealed.

	Modulus of elasticity. Direct	Shearing	
0.05% C........................	26,000,000	13,000,000	
0.10% C........................	28,000,000	13,500,000	
0.25% C........................	30,000,000	14,000,000	
0.40% C........................	30,000,000	14,000,000	
0.90% C........................	32,000,000	14,500,000	When well annealed.
1.00 to 2.00% C................	35,000,000	16,000,000	When hardened.

	Modulus of Rupture	
For flat plates..	1.0	(t. s.)
For squares...	1.2	(t. s.)
For high rectangles...	1.5	(t. s.)
For rounds and diamonds...	1.8	(t. s.)

Strength of Steel Castings—Well Annealed—

Small Castings

Tension Ultimate	yield point	Compression Ultimate	yield point	Shearing Ultimate	yield point
60,000	30,000	80,000	45,000	45,000	25,000

Modulus of elasticity (direct).. 29,000,000
Modulus of elasticity (shearing).. 13,000,000
Modulus of rupture in same ratios as above.

Large Castings

Tension Ultimate	yield point	Compression Ultimate	yield point	Shearing Ultimate	yield point
40,000	20,000	70,000	40,000	40,000	20,000

Modulus of elasticity (direct).. 28,000,000
Modulus of elasticity (shearing).. 13,000,000
Modulus of rupture in same ratios as above.
Steel Castings with Vanadium show about 20 per cent increase over the above values.

Special Steel Alloys—Pounds per Square Inch

	Tensile strength.			Yield point
Manganese........................	Up to	140,000	up to	90,000
Nickel............................	"	140,000	"	90,000
Chrome Vanadium.................	"	200,000	"	175,000
Chrome Nickel....................	"	140,000	"	100,000
Tungsten..........................	"	170,000	"	150,000
Chrome Tungsten.................	"	185,000	"	160,000

Strength of Copper, Aluminum and Various Alloys—Pounds per Square Inch

Cast Copper

Tension..	22,000
Compression......................................	45,000
Shearing...	18,000
Modulus of elasticity (direct)...................	12,000,000
Modulus of elasticity (shearing).................	8,000,000
Modulus of rupture (rectangular sections)........	35,000

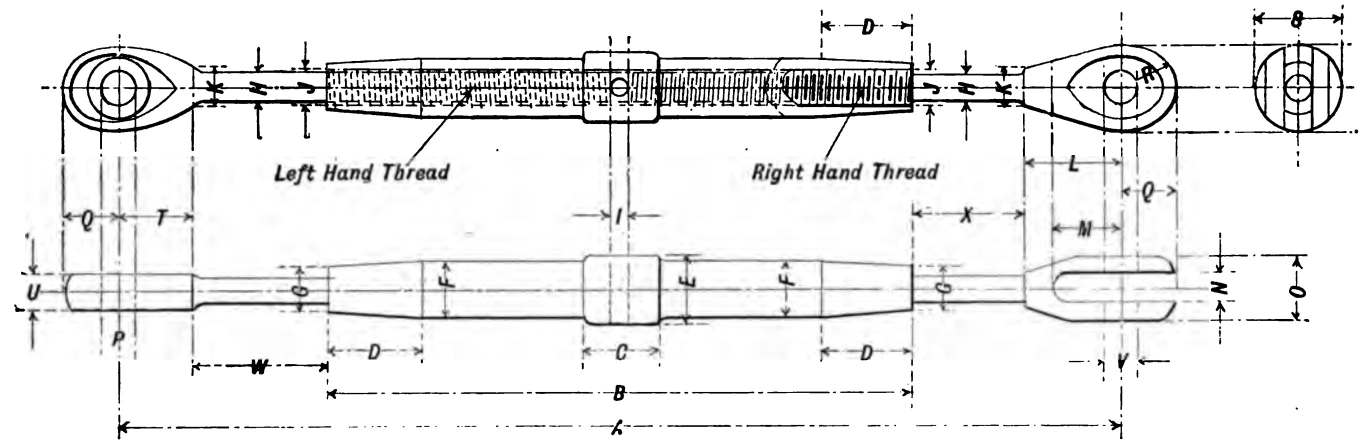
Cold Rolled or Hammered Plates

Tension..	32,000
Compression......................................	60,000
Shearing...	28,000

Cold Drawn Wire

Tension..	50,000 to 60,000
Shearing...	40,000

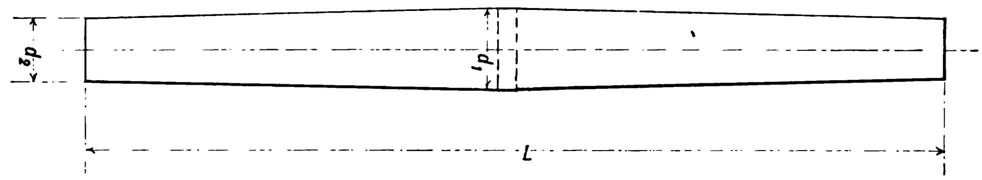
Modulus of elasticity of cold worked copper 17,000,000. Copper has no well defined yield point.

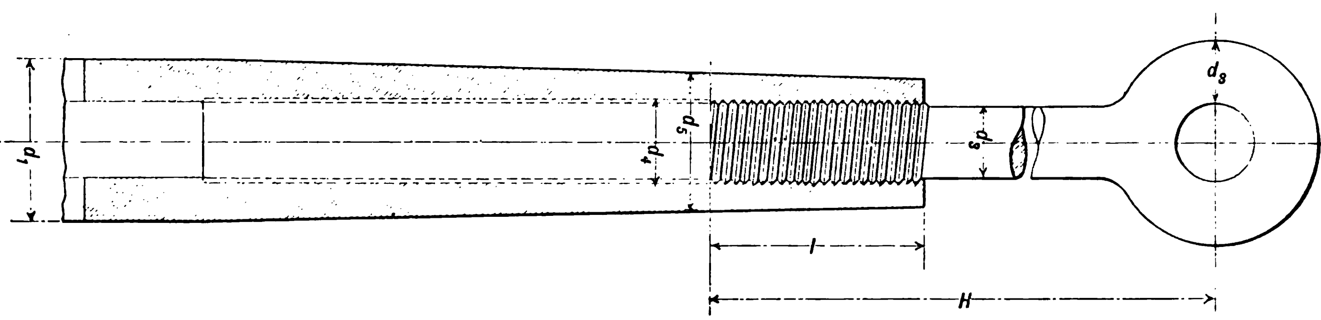


Curtiss No.	No.	A	B	C	D	E	F	G	H	I	J	K	L	M	N	O	P	Q	R	S	T	U	V	W	X	Th'ds Per Inch	Br'k'g Stress	Cable	Weight
Special		7 3/4	4	1/2	7/8	5/8	5/16	11/16	.320	3/32	3/8	3/8	7/8	5/16	11/64	11/32	5/16	1/2	5/16	11/16	5/8	21/64	5/16	1 1/8	1 1/8	24	8846	1/4 in. Cable	...
3	326	8	4	7/16	5/16	3/8	21/64	5/32	.159	1/8	.203	1/4	5/8	1/2	.110	5/16	3/16	21/64	3/16	1/2	7/16	3/16	3/16	1 1/2	1 15/32	30	2183	3/64	...
4	327	7 11/16	4	1/2	5/8	23/64	29/64	5/16	.187	3/32	.234	3/32	11/16	1/2	11/64	7/16	1/4	21/64	3/16	5/8	1/2	3/16	1/4	1 5/8	1 11/32	28	3037	1/8 in. Cable	...
5	328	7 13/16	4	1/2	11/16	1/2	7/16	3/8	.215	3/32	.265	5/16	13/16	5/16	11/64	7/16	1/4	21/64	1/4	11/16	1/2	1/4	1/4	1 5/16	1 11/32	26	3993	5/32	...
6	329	7 11/16	4	1/2	3/4	11/32	21/64	7/16	.258	3/32	5/16	3/8	7/8	5/8	11/64	5/16	5/32	11/32	5/32	3/4	5/16	5/32	5/32	1 3/16	1 3/16	24	5750	3/16	...
	326-S	3 13/16	2	7/16	5/16	3/8	21/64	5/32	.159	1/8	.203	1/4	5/8	1/2	.110	5/16	3/32	21/64	3/16	1/2	7/16	3/16	3/16	7/16	15/32	30	2183	10GA Wire	...
	327-S	4 5/16	2 1/8	1/2	5/8	23/64	21/64	5/16	.187	3/32	.234	3/32	13/16	5/8	.110	11/32	1/4	21/64	3/16	5/8	1/2	3/32	1/4	7/16	15/32	28	3037	8GA Wire	...

Material:—Mang. Bronze and 55-Ton Steel

FIG. 1.— CURTISS TURNBUCKLES.





SPECIMEN	L	H	d_1	d_2	d_3	d_4	d_5	l	Weight lbs.	Estimated Load. Pounds	Remarks
National No. 324	2.35	1.30	.28	.23	.11	.15	.20	.45	.039	1250	Failure in shank.
National No. 327	2.50	2.50	.40	.32	.19	.23	.35	.60	.161	2200	Yield in barrel.
National No. 328	5.00	2.75	.48	.37	.20	.27	.40	.65	.229	2800	Yield in barrel.
National No. 328	5.00	2.80	.46	.38	.22	.27	.40	.70	.242	2200	Low value due to d_1
National No. 329	6.00	3.25	.54	.44	.23	.31	.47	.85	.361	4000	Yield in barrel.
A. J. Meyer No. 3	2.35	1.30	.28	.21	.11	.15	.19	.45	.037	1200	Yield in barrel. Above average.
A. J. Meyer No. 4	3.00	1.75	.32	.20	.14	.18	.28	.45	.0682	1500	Yield in barrel.
A. J. Meyer No. 6	4.50	2.50	.42	.32	.19	.23	.35	.55	.161	2200	Yield in barrel.
A. J. Meyer	6.00	....	...	...	.20	...	...	...	.172	2800	Failure in shank.
A. J. Meyer	4.80	2.70	.62	.51	.30	.33	.57	.80	.506	7000	Yield in barrel.

Estimated load based upon average strength at yield point of barrel 41,000 lb. per sq. in. Average tensile strength barrel, 60,000 lb. per sq. in. Average tensile strength of shank 130,000 lb. per sq. in.

FIG. 2:—MEYER, NATIONAL, BURGESS AND BINET TURNBUCKLES.

CAST ALUMINUM

Tension	14,000
Compression	25,000
Shearing	10,000

COLD WORKED ALUMINUM PLATE AND WIRE

Tension	24,000 (Plate)
Tension	40,000 (Wire)
Modulus of elasticity (direct)	8,000,000

Aluminum has no well defined yield point.

DURALUMIN

Tension, 33,000; Compression, 68,000; Modulus elasticity, 10,000,000.

High brass	Tension	35,000
Low brass	"	28,000
Composition	"	30,000
Bronze	"	30,000 to 75,000 varying with composition.
Phosphor bronze	"	40,000 to 130,000, varying with form.
Tobin bronze	"	80,000, yield point 50,000 hot rolled.
Tobin bronze	"	100,000, yield point, 70,000 cold rolled.
Delta metal	"	45,000 Cast. Tension, 70,000 cold rolled.
Manganese bronze	"	70,000 lbs./sq. in. Compression, 120,000 lbs./sq. in.
Monel metal (hot rolled)	Tension	(ultimate) 88,150 lbs./sq. in.
	"	(yield. 58,000 lbs./sq. in. elongation in 2 inch, 38 per cent.

The steel at present in common use among manufacturers is the mild sheet steel generally designated as cold rolled steel (C.R.S.). Whether its easy working qualities or its cheapness and easy supply has brought about this poor choice, is hard to judge. It is a relief to find that the constructors and the Government are endeavoring to do away with this most unreliable and inefficient of materials. Instances have only too often been brought to our attention when upon the completion of some small stamped fitting, the apparently solid metal is found to be of two distinct layers of thinner metal held together only at a few points. It is not well, however, to jump too quickly to the other extreme and attempt to use the very high strength alloys—requiring expert working and heat treatment. Companies to-day are, with a few exceptions, not in a position to undertake this added responsibility and the forcing of such delicate work upon inexperienced hands would be as dangerous as the present methods.

The following table outlines the general influence of chemical composition of the physical properties of steel:

	Tensile strength.	Elastic ratio.	Ductility.	Hardness	Hardening power	Brittleness.	Resistance to shock.	Resistance to vibratory stresses.	Forgeability, absence of red-shortness.	Weldability.
Carbon	(1)	+	—	+	(2)	+	(12)	+	—	—
Phosphorus	(13)	...	(3)	0	...	(4)	—	—	0	...
Manganese	+	—	...	+	...	...	...	...	(5)	...
Sulphur	...	...	...	0	...	...	...	...	—	—
Silicon	...	...	(6)	0	...	...	...	...	(7)	...
Aluminum	...	...	(8)	0	...	...	...	...	...	—
Vanadium	+	+	(9)	...	...	...	+	+	...	...
Chromium	+	+	...	+	...	...	+	+	...	...
Molybdenum	+	...	...	+	...	...	...	...	...	...
Tungsten	+	+	...	+	...	...	...	...	...	...
Nickel	+	+	(9)	+	...	...	+	+	...	...
Copper	...	...	...	0	...	...	...	...	...	(10)
Oxygen	...	...	...	...	...	+	...	...	(11)	...
Nitrogen	...	...	...	...	...	+	...	...	...	...

(1) + up to 1 per cent., then —; (2) + up to 0.85 per cent.; (3) 0 up to 0.1 per cent., then —; (4) + over 0.1 per cent.; (5) + up to 0.5 per cent.; (6) — above 1.5 per cent.; (7) — over 6 per cent.; (8) — over 0.85 per cent.; (9) — slightly; (10) — over 1 per cent.; (11) — from 2 to 7 per cent.; (12) + up to 0.3 per cent., then —; (13) + up to 0.1 per cent., then —.

The above table, while very comprehensive, should not be considered as final.

The matter of weldability of the chrome vanadium and nickel steel is indefinite, very reliable information showing that 3¼ per cent. nickel steels give better welds than the chromium steel.

The S.A.E. Specification No. 3130 for a low carbon chrome-nickel steel, or Specification No. 2330 for 3½ nickel steel would seem to meet the requirements of the manufacturers as well as the Army Specifications of the S.A.E. No. 6130 chrome-vanadium, eliminating at the same time, the great danger of segregation due to faulty heat-treatment.

Heat treatment and its influence cannot be gone into; it requires very careful study and infinite care in application.

Strength and Weight of Mild Steel Rivets and Pins

Diameter, inches. "d."	Strength in Pounds Single Shear.	Double Shear.	Crushing + t.
1/8	1,000	1,600	1,200
3/16	1,600	3,000	1,800
1/4	2,750	4,500	2,400
5/16	3,900	7,000	2,800
3/8	5,000	9,800	3,600
1/2	9,000	17,800	4,800
3/4	20,000	40,000	7,200
1	35,500	71,000	9,600

The above values are based upon a shearing strength in pounds per square inch $fs = 45{,}000$ and a crushing strength, $f_c = 96{,}000$.

$$P_s = f_s \frac{\pi\, d^2}{4} \text{ for single shear.}$$

$P_c = f_c dxt$, for crushing, where t, the thickness of the plate or other piece held by the rivet or pin.

The values for crushing have been worked out for a plate 1 inch thick, therefore the crushing strength for various thicknesses of plate may be computed by multiplying the above values by "t" in inches.

If the crushing strength of the rivet is greater than its shearing strength the design should be based upon the smaller result.

In case the bolt is subjected to a load other than tension, the strength should be based upon the corresponding form of loading. When bolts and pins are used in turnbuckle fittings and wire connections they are usually subjected to a form of bending and should be calculated as a beam round cross section loaded at the center.

$f = \frac{M_y}{I}$, where f = modulus of rupture, M = bending moment due to load generally considered concentrated at the center, Y = distance from neutral axis to most strained fiber, in this case ½ D; I = moment of inertia of cross section.

In figuring bolts that pierce wooded members the failure of the wood should be considered first, since in this type of connection rupture is most often caused by the fastening pulling out or loosening due to the wood crushing in front of the $P = f_c \times L \times D$. Where P = crushing load, f_c = crushing strength of wood, L = length of bolt in wood, D = normal bolt diameter.

United States Standard Bolts and Nuts

Nominal dia. D., in.	Eff. dia. root of thread d, in.	Shank area, sq. in.	Eff. area root of thread A, sq. in.	Threads per in.	Short dia. of hex. and sq. nuts, in.	App. dia. across corners hex. nuts, in.	App. dia. across corners sq. nuts, in.
1/4	0.185	0.049	0.027	20	1/4	0.578	0.707
5/16	0.240	0.077	0.045	18	19/32	0.686	0.840
3/8	0.294	0.110	0.068	16	11/16	0.794	0.972
7/16	0.345	0.150	0.093	14	25/32	0.902	1.105
1/2	0.400	0.196	0.126	13	7/8	1.010	1.237
9/16	0.454	0.249	0.161	12	31/32	1.119	1.371
5/8	0.507	0.307	0.202	11	1- 1/16	1.227	1.503
3/4	0.620	0.442	0.302	10	1- 1/4	1.443	1.768
7/8	0.731	0.601	0.420	9	1- 7/16	1.661	2.033
1	0.837	0.785	0.550	8	1- 5/8	1.876	2.298
1-1/8	0.940	0.994	0.694	7	1-13/16	2.093	2.564
1-1/4	1.065	1.227	0.891	7	2	2.310	2.828
1-3/8	1.160	1.485	1.057	6	2- 3/16	2.527	3.094
1-1/2	1.284	1.767	1.295	6	2- 3/8	2.743	3.358
1-5/8	1.383	2.074	1.515	5-1/2	2- 9/16	2.959	3.624
1-3/4	1.431	2.405	1.746	5	2- 3/4	3.176	3.889
1-7/8	1.616	2.761	2.051	5	2-15/16	3.393	4.156

NOTE:—Depth of nut = nominal diameter.

Depth of head = one-half short diam. of hex. and sq. nuts.

The strength of United States standard bolts may be based upon the formula $P = A \times f_t$. Where P = the strength of the bolt, f_t = tensile fiber stress per square inch = 65,000, A = effective area at root of thread. Solving for D the nominal diameter $D = 1.24 \sqrt{\frac{P}{f_t}} + .088$.

Airplane Fabrics

The general requirements and tests for airplane fabrics are well summarized by the following table:

(1) Fabric should present reasonably great resistance to flame.
(2) It should be proof against the action of salt water, moist air, extreme dryness quick changes of temperature.
(3) It should not stretch in any direction.
(4) It should have a tensile strength of at least 75 lb. per inch width in any, direction.
(5) The tendency to tear and split because of tacks, bullets, etc., should be almost nil.
(6) The weight should be taken in an atmosphere of 65 per cent relative humidity at 70° Fahr.
(7) The weight, yarn number and tensile strength of the fabric should be obtained when it is in a bone dry condition, i. e. after it has been subjected to a temperature of 221° Fahr. for two hours.
(8) Identity and average length of fibers should be ascertained.
(9) Determination should be made of the percentage moisture "regain" under the available range of temperature and humidity.
(10) A shrinkage determination should be made.

Some Representative Specifications, Strength and Weight Figures

In Table 4 are given some representative specifications which represent the average values of current practice.

TABLE 5.

Specifications	Weight per square yard ounces	Threads per inch warp	Threads per inch weft	Strength per inch width (warp) lb.	Strength per inch width weft (lb.)
Curtiss No. 66........ Oct. 26, 1916	4	96	100	91	102
R. A. F. No. 17-C.....	4	92	95	92	95
U. S. Army No. 1002..	3.75–4.4	94	100	75	85 (minimum)
McBratney tests......	4	95	99	—	108
Clarence Whitman (cotton)............	4	275 threads per sq. in.		87	96

Messrs. Lamb, Finlay and Co. have kindly communicated the average values of tests extending over a number of shipments, in accordance with British Government specifications:

"Three pieces are cut from the length and three pieces from the width of the goods. These samples must be long enough to leave 6 in. clear between the grips and sufficiently wide to leave the test pieces 2 in. wide after trimming. The test pieces are soaked in water for two hours. They are then taken out and the excess water is removed and the goods put in an Avery machine, the load being applied at the rate of 50 lb. per inch width per minute."

This method of testing, while apparently arbitrary, is convenient and avoids errors due to humidity changes, and is preferable to a test on dry material.

In the National Advisory Report the following figures are given for fabrics of diff. ent weights:

	Weight in ozs. per sq. yd.	Strength Warp	Strength Filler
I.	3.67	65.0	54.4
II.	3.78	69.5	49.2
III.	3.87	80.7	79.0
IV.	4.04	86.9	74.0
V.	4.09	90.2	82.7
VI.	4.48	82.9	100.1
VII.	4.60	95.0	60.0
VIII.	4.86	90.4	102.5

TABLE 6.

Quality	Width inches.	Approximate breaking strength per inch. Warp.	Weft.	Weight per sq. yd.
A24....................	36	72 lb.	78 lb.	4.00 os.
A29....................	36	90.5 "	101.9 "	4.00 "
A31....................	36	108.5 "	94.6 "	4.00 "
A33....................	36	114.7 "	103.6 "	4.00 "
*A37....................	36	112 "	115.6 "	4.12 "
*A40....................	36	129.4 "	123.4 "	3.75 "

*British Government Standard qualities.

Wing Dope and Varnish

The following notes on dope and varnish may be of interest to the manufacturer:

Dope alone on Irish linen surfaces has proven very satisfactory. Four to five coats, allowing about one-half hour for drying between each coat are ample protection for the most severe conditions. Very rigorous tests on samples exposed to the weather during the month of February resulted in no ill effects, the cloth remaining tight, glossy and without spots, cracks or tears. The weight of the covering is increased about .66 oz. per square yard per coat, with an application of approximately one gallon to ten square yards.

Varnish finish is recommended in many cases as more permanent and, being less effected by salt water, has some advantages on water machines. When repairing is to be done it is first necessary to remove the varnish before the patch is applied with dope, as a glue, causing some inconvenience.

Doped surfaces have about 8 per cent to 10 per cent more strength and more resistance to tearing. It is necessary to redope all surfaces every three to five months.

Cotton and silk fabrics have a tendency to rot when covered with dope or varnish and such surfaces are not recommended.

References, Part II, Chapter 5

AIRPLANE FABRICS

"First Annual Report of the National Advisory Committee on Aeronautics."
"Balloon and Aeroplane Fabrics," by Willis A. Gibbons and Omar H. Smith.
"Circular No. 41." *Bureau of Standards.*

TIMBER

Judge's "Aeroplane Design."
"Material of Construction," Adelbert P. Mills.
"Mechanical Engineers' Handbook," Lionel S. Marks.
"Lanza's Applied Mechanics."
"The Mechanical Properties of Wood," by Samuel J. Record (containing numerous references.)
"Reports of Tests on the Strength of Structural Material," *Made at the Watertown Arsenal, Mass.*
Publications of the U. S. Forest Service on "The Mechanical Properties of Wood and Timber Testing."

METALS

"Materials of Construction," Prof. Upton, of Cornell University.

WIRES AND CABLES

First Annual Report of the National Advisory Committee on Aeronautics. Aviation Wires and Cables."
(This also contains much valuable information on fastenings.)

TURNBUCKLES

Arthur Orr, AVAITION AND AERONAUTICAL ENGINEERIEG, December 1, 1915.

Chapter VI

Worst Dynamic Loads; Factors of Safety

One of the most difficult problems in aeronautics is the estimate of the worst loads likely to come on under unusual circumstances, on which alone correct allowances for factors of safety can be based. In speaking of factors of safety, a distinction must be made between the load factor of safety and the gross factor of safety. Thus, if the load factor of safety for a certain part of the machine is four, the material employed may be so untrustworthy that an allowance for it, of say one and a half, may have to be made, bringing up the gross factor of safety to six. It is the gross factor of safety which is commonly spoken of as the factor of safety.

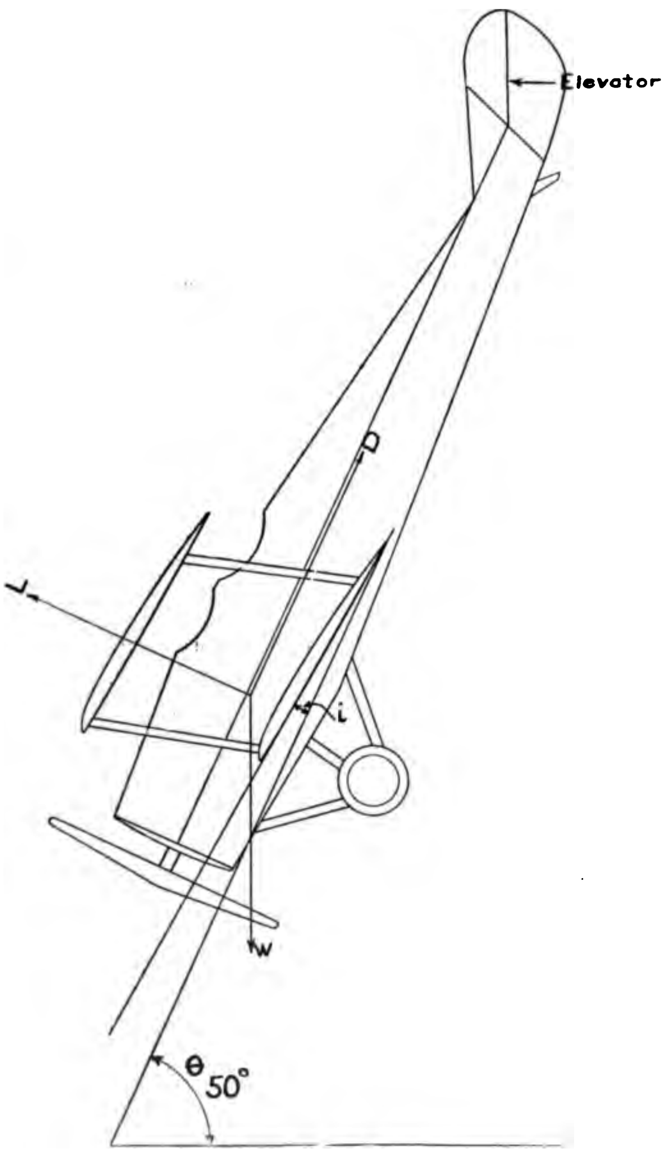


Fig. 1

Conditions Under Which Heavy Loads Come

Heavy loads come on an airplane under so many conditions, that the following classification is probably incomplete. It is, however, all that is possible in the present stage of the art.

(A) In the air:

(1) in flattening out of a steep dive
(2) in heavy banking
(3) in looping
(4) in sudden gusts.

(B) On the ground:

(1) on landing

We shall consider these conditions one by one as far as possible.

The wing structure will meet with the greatest loads in the air. On landing, the wing structure has to meet only the shock resulting from the decceleration of its own weight which may be some 12-15 per cent of the weight of the machine, while in the air, it has to support the whole weight of the airplane and under certain conditions five or six times the whole weight. It is in the air, therefore, that the wings meet the worst conditions. The body may carry severe stresses in the air when powerful forces come into play on the rudder and elevator, but it may also be powerfully stressed on landing. For chassis design, it is only landing and taxi-ing stresses that need be considered.

(1) Flattening Out After a Steep Dive

The exact mathematical computation of stresses in such a case is not yet possible; it is, however, interesting to see how such stresses arise, and how they are limited.

In order to have a concrete case, we will consider the Clark model tested at the Massachusetts Institute of Technology, and described in Hunsaker's "Dynamical Stability." This had the following estimated characteristics:

Wing area including ailerons	464 sq. ft.
Span	40.2 ft. mean
Area, stabilizer	16.1 sq. ft.
Area, elevators	16.0 sq. ft.
Area, rudder	9.35 sq. ft.
Length, body	24.5 ft.
Weight (tanks half full)	1600 lb.
Radii of gyration	5.2 ft. in roll 4.65 ft. in pitch 6.975 ft. in yaw
Brake horse-power	110
Maximum speed	87 m.p.h.
Minimum speed	35 m.p.h.
Best glide	1 in 9

For a tail setting of —5 deg. to the wings, the model (1/26th full size) had the following forces acting on it at a speed of 30 m.p.h., which we shall use without correction for transferring to the full size machine:

Angle	Lift on model at 30 m.p.h.	Drift on model at 30 m.p.h.
—4	—.115	+.128
—2	+.112	+.108
—1	+.240	+.104
0	+.360	+.101
+1	+.490	+.102
+2	+.625	+.105
+4	+.872	+.115
+8	+1.305	+.153
+12	+1.568	+.213
+16	+1.640	+.370
+18	+1.580	+.498

In the U. S. Army Specifications 1002 (reprinted in AVIATION AND AERONAUTICAL ENGINEERING of November 1, 1916), one of the stipulations for airworthiness is that the pilot may be required to dive at an angle of 50 deg. to the horizontal, to maintain such a dive for one or two seconds, and then to pull out reasonably quickly. We will assume that the dive is continued for even a longer period so that the limiting velocity is reached, and then try to see what will happen on flattening out sharply.

The propeller thrust on the dive may be neglected whether the engine is cut off or not, the slip being so enormous as to reduce it to a negligible quantity.

Considering the sketch of Fig. 1, the equations of motion evidently are

$$D = W \sin \theta$$
$$L = W \cos \theta$$

The most straightforward way of finding at what incidence to the flight path the machine is under steady limiting conditions, is one of trial and error. After one or two trials, we find that the angle of incidence, 2¼ deg. will satisfy conditions.

The drag on the model at this angle is 0.111 lb., and the lift 0.094 lb. at 30 m.p.h. The two equations are very nearly satisfied. Thus converting to full size conditions:

(1) $W \sin 50 \text{ deg.} = (1600)\ (0.7660) = 0.1235 =$

$$\frac{0.111 \times 26^2}{30^2} V^2 \text{ and } V^2 = 14{,}850,\ V = 122 \text{ m.p.h.}$$

(2) $W \cos 50 \text{ deg.} = (1660)\ (0.06428) = 1030$ while

$$\frac{0.094 \times 26^2}{30^2} \times 0.14850 = \text{Lift} = 1040 \text{ lb.}$$

the difference of 10 lb. in the lift being negligible.

If, at this point, the pilot throws his elevator hard up, he will increase his angle of incidence rapidly, and move his path more and more to the horizontal. The rapidity with which he can come out of the dive depends on the force which he can bring to bear on the elevator, and is resisted by the inertia of the machine, and the damping against angular rotation. There is reason to believe that during this process he loses very little speed. The equations of motion during this process are somewhat complicated and cannot be solved directly. But if we assume that for a machine of this type, the pilot can change his angle of incidence to say 8 deg. without losing speed, the lift on the model at this speed being 1.305, the lift on the machine becomes

$$\frac{1.305}{30^2} \times 26^2 \times 122^2 = 14{,}400 \text{ lb.}$$

or a load of nine times the weight of the machine. It is commonly accepted that the actual load is not quite so great, being between 5 and 6. The pilot could not easily wreck a machine with moderately strong controls, and weights distributed far from the center of gravity giving a large moment of inertia. But with a light machine, with weights close to the center of gravity and a powerful elevator, a reckless recovery would be highly dangerous.

It should be pointed out that the uncertainty as to the exact movements a machine goes through on flattening out, makes the question of the angles of incidence at which loads on front and rear spars should be distributed and computed a very controversial one. The latest U. S. Army specifications call for a stress diagram at 15 deg., which throws the greater load on the front spar. If, as is quite possible, a machine flattening out after a steep dive does not reach such a high angle of incidence, but arrives at some intermediate angle such as the 8 deg. mentioned above, then it would be fairer to draw a stress diagram at this angle of incidence, with a more equal distribution between the two spars.

(2) Loading in Heavy Banking

The loading on a steep bank is dependent on the speed, radius of turn, and angle of bank.

In the sketch, Fig. 2, the machine is moving out of the plane of the paper and turning at an angle of bank θ. The three forces acting in the vertical plane of the machine are the lift, the weight and the centrifugal force, which may be assumed as acting at the center of gravity of the machine. Resolving L along the lines of these two forces, we have as equations of equilibrium

$$L \sin \theta = K_y\, AV^2 \sin \theta = \frac{W}{g} \frac{V^2}{R}$$
$$L \cos \theta = K_x\, AV^2 \cos \theta = W$$

where V = speed in feet per second, and R = radius in feet. From these equations, one important fact appears, that on a

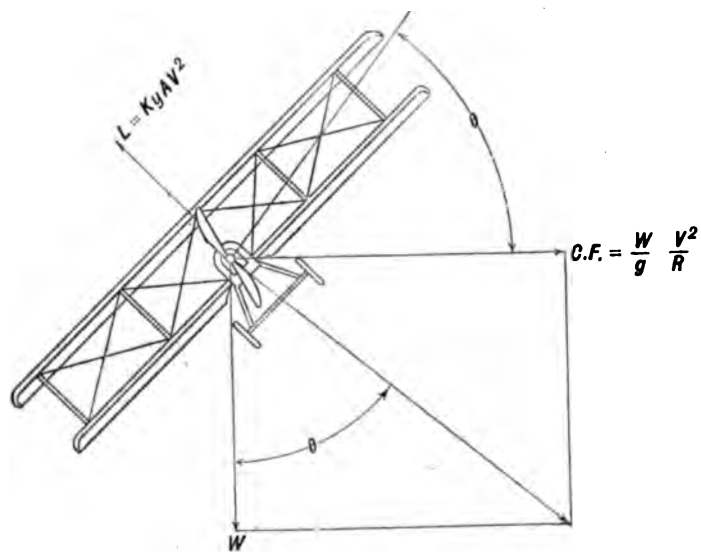


FIG. 2

steep bank where $\cos \theta$ is small, the lift of normal flight is insufficient, and that before banking a pilot must increase his power and speed, otherwise his machine may drop on the bank.

The load on the machine in banking will increase with the centrifugal force to be overcome in addition to the weight, and is, therefore, greatest when the velocity has increased beyond the maximum in normal flight and when the radius of turn is very small.

For the Clark model previously considered, we will assume that the machine has attained a speed of 120 m.p.h. or 176 ft. per second, after a dive and that the pilot goes into a sharp turn of 400 feet radius.

From the equations of equilibrium we have

$$\tan \theta = \frac{\frac{W}{g} \frac{V^2}{R}}{W} = \frac{V^2}{gR} = \frac{30.500}{32.2 \times 400} = 2.37$$

$\theta = 67$ deg. and $\sec \theta = 2.559$, which is certainly a fairly steep angle of bank. Since $L \cos \theta$, $L = \sec W = 2.559\ W$.

It is possible to consider a case where the velocity would be still greater than the 120 miles per hour, and the radius still smaller, in which case the loading might still be heavier. It does not seem probable, however, that the worst possible loading on a bank would exceed 3 or 4 W.

The angle of incidence on a bank interests us again from the point of center of pressure and distribution of pressure between the two spars. Considering the Clark model of the previous paragraph, $W = 1600$ and $L = 2.559$ $W = 4100 = K_y \times 464 \times 120^2$

from which $K_y = \dfrac{4100}{464 \times 14{,}400} = 0.000615$

and the angle of incidence is not much above 0 deg. for the Clark machine on such a bank.

(3) Loading in Looping

In looping similar methods would be employed as in considering flattening out after a dive. The probable maximum loading is estimated to be 4.

(4) Stresses Due to Gusts

Another cause of violent stresses is in the action of sudden gusts on a machine, where the inertia tends to maintain the same speed for a different angle of incidence, or the same incidence for a different speed.

The machine may encounter:

(a) a head-on gust
(b) a following gust
(c) an up-gust
(d) a down-gust.

Granted a sufficiently violent gust, there is no possible limit to the stresses which may come on a machine in such cases, and a hurricane might wreck a machine for whatever factor of safety it was designed. It is necessary to investigate, however, whether the gusts, as we may expect to occur in ordinary practice from our metereological data, are well within safe limits.

(a) Head-on Gusts

Imagine the Clark machine to be moving at 59 m.p.h. at an angle of incidence 2 deg. against a head-on wind of 20 m.p.h. so that its absolute velocity relative to the earth is 39 m.p.h. If the head-on wind increases to 30 m.p.h., the absolute velocity relative to the earth will still remain at 39 miles for a second or two. During this period, the velocity to the air will increase to 69 miles per hour, with the angle of incidence unchanged. The lift on the machine will, therefore, be momentarily increased in the ratio of $\dfrac{69^2}{59^2} = 1.36$.

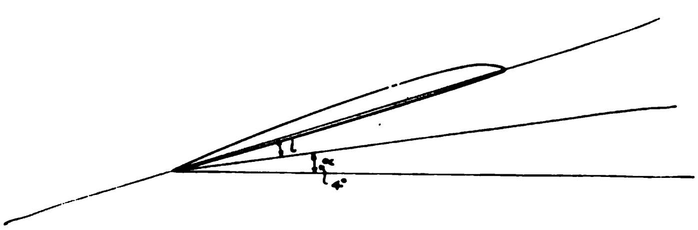


FIG. 3

There will be an acceleration upwards and an increased load on the machine $= 1.36\ W$.

(b) Following Gusts

If the machine were traveling in a following wind, which suddenly diminished, a similar action would ensue, since the relative velocity to the air would here also increase.

If, on the other hand, in the case considered above the head-on gust suddenly diminished to 10 miles an hour, the relative velocity to the air would be decreased to 49 miles per hour, and the lift would be diminished in the ratio of $\dfrac{49^2}{59^2} = 0.69$ and the lift on the wings would, in this case, be actually less than in normal flight, so that the machine would drop.

It is also clear from the above that the gust effects are most important, when the speed of the machine is lowest.

(c) Up-gusts; (d) Down-gusts

Without going into numerical examples, we can see easily from Fig. 3, the effect of an up-gust in increasing the load. The up-gust both increases the velocity of the relative wind and its angle of incidence, with a corresponding increase in lift, except at very high angles where a reverse effect is possible. For a down-gust the converse would hold true.

With normal flying weather, the effect of gust should never increase the load to more than two or three times the weight of the machine.

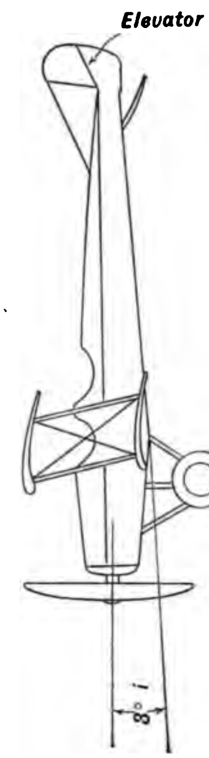


FIG. 4

Limiting Velocity for a Sheer Vertical Dive

A sheer vertical dive is unlikely to occur and is not required for stress calculations in practice, but it is interesting to note the extreme limiting velocity in such a case. A sheer vertical dive is only possible if the machine is at the angle to the vertical which gives no lift, and the elevator is set only at such an angle that the moment of the total drag about the center of gravity is neutralized.

For the machine in question the angle of no lift is —3 deg.

The drag on the model at this angle is 0.118 lb. To find the limiting velocity, we can write

$$W = 1600 = \frac{0.118 = 26^2}{30^2} \times V^2$$

from which $V = 134$ m.p.h., which is not so very much greater than the limiting speed on the 50 deg. drive.

Worst Loads on Landing

The computation of such loads is connected with chassis design, and we shall deal more fully with it later. Some calculations taken from "Notes on Aeroplane Shock Absorbers of Rubber" are an interesting introduction to the subject:

An airplane weighing W pounds striking the ground at V feet per second on a glide of 1 in 7 has kinetic energy to be absorbed by the landing gear of $\dfrac{W}{2g}\left(\dfrac{V}{7}\right)$ If the machine comes to rest after a motion of x feet, the work done by gravity on it is Wx, and the total stored in the shock absorber is $W\left\{x + \dfrac{1}{2g}\left(\dfrac{V}{7}\right)^2\right\}$. The average force in the springs is half the maximum F, given by the equality:

$$\tfrac{1}{2}Fx = W\left\{x + \frac{1}{2g}\left(\frac{V}{7}\right)^2\right\}$$

or

$$F = W\left\{2 + \frac{1}{xg}\left(\frac{V}{7}\right)^2\right\}$$

If we take ordinary conditions as $V = 66$ ft. per sec. (45 m. p. h.) $F = W\left(2 + \dfrac{2.77}{x}\right)$, from which we get the following table for use in design:

x	F
1 in.	35.0 W
2 "	19.0 W
3 "	13.0 W
4 "	10.3 W
5 "	8.6 W
6 "	7.5 W
8 "	6.1 W
10 "	5.3 W
12 "	4.8 W

It appears that the load on the landing gear is nearly 14 times the weight of the airplane, if a motion of only 5 in. is allowed. This requires an excessive factor of safety and makes a very heavy construction. Of course, no allowance has been made for the collapse of pneumatic tires which may add 2 in. to the motion of the recoil mechanism.

References—Part II, Chapter 6

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Chapter VII

Preliminary Design of Secondary Training Machine

Preliminary Weight Estimates

Every designer will approach the design of a new machine in a different manner, and no definite rules can be laid down. In the design of a standard secondary training machine, we have the advantage of following well-known lines. with such excellent examples of machines tried out in practice as, for example, the Curtiss JN. Following such practice we can make very close estimates of possible weights and performances. and easily determine possible wing and control surfaces. A new and difficult type, such as a military twin-hydro, would require long preliminary study.

Recalling Army Specification No. 1001—detailed in Part 2, Chapter 1—we have to meet the following requirements:

Pilot and passenger, 330 lbs.
Gasoline and oil for 4 hours' flight
Engine between 90 and 110 h.p.
Maximum speed, 75 m.p.h.
Minimum speed, 43 m.p.h.
Climbing speed, 3000 ft. in 10 min.
Two wheel landing gear

From the engine data of Chapter 5 we could select a number of suitable engines. We shall select the Curtiss 90 h.p. OX. It is with this engine that a student has designed a similar machine, the salient features of which we shall embody in the post-graduate course at the Massachusetts Institute of Technology.

Practice shows that the above performances can be achieved with a weight of about 1850 lb., i.e., 20.4 lb. per horsepower, and we shall make this our preliminary estimate. This figure is slightly less than that of the JN-4, but is probably very near to the JN-4B.

The first step is to set down all weights of which we can be fairly certain, and on which no improvement is possible, thus:

		lb.
A B	Pilot and passenger in aviation dress	330
C	Engine and accessories	360
D	Radiator	50
E	Water in engine and radiator and piping	40
F	Propeller and hub	35
G	Gasoline tank of 40 gallons capacity	30
H	Gasoline and oil for 4 hours' flight	220
I J	2 Instrument boards, with a set of barograph, tachometer, air-speed indicator and clock on each	40
K L	2 Dep controls	25
		1130

This leaves us with some 720 lb. available for the purely structural parts of the machine: chassis, complete body assembly, wings, interplane bracing, and tail surfaces. On the Curtiss JN-4 (Part 2, Chapter 3) we have the following percentages for these groups:

Chassis	4.03%	equivalent in our machine to	74.5 lb.
Wings	14.15%	" " " " "	261.0 "
Interplane bracing	4.95%	" " " " "	91.5 "
Tail surfaces	2.76%	" " " " "	51.0 "
Body assembly	15.55%		
equivalent to 286.0 lb., from which must be deducted 40 lb. for instrument boards and instruments, leaving			246.0 "
			724.0 "

Since we are following standard practice very closely we can take the above figures to hold fairly well for various parts of the machine.

Choice of Wing and Area

For a machine of this type it is not necessary to have a wing of extreme characteristics. It is more practical to select a good all-round wing, with fair structural characteristics, than to choose a wing with high efficiency at low speeds, but a low lift coefficient at maximum angles, and the R.A.F.6 can be adopted without much chance of mishap.

In a machine of the pursuit type, it would be worth while trying a number of different wing areas, but in a training machine it is, in the first place, essential to secure the necessary landing speed, and then to attain as high a speed and climb as possible with careful design. It only remains for us to find the maximum K_y of the R.A.F.6 and the necessary correcting factor for biplane effects.

There is, first of all, the question of stagger to be considered. The increased efficiency due to staggering is offset by questions of weight and head resistance, while the increase in efficiency is not so very important. Stagger is, therefore, mainly determined by considerations of the view obtainable by pilot or passenger. On this particular machine we shall employ a very slight stagger of about 5 per cent, giving a good overhead view for the pilot.

Overhang is likely to improve efficiency, but no aerodynamical data is available. It must be remembered that a large overhang, together with the aileron loads, imposes a very serious load on the rear spar of the wing. If any unsupported overhang is employed, it should be less in length than the gap. We should, in the present state of the art, make no improvement in the K_y correction for biplanes on account of overhang.

We must also settle on gap/chord ratio. We have seen in our aerodynamical work the improvement consequent on great gap/chord ratio. But to offset this, we have the question of increased weight and resistance of struts and wires. For triplanes with their blade-like wings, a very high gap/chord ratio is, no doubt, permissible, but for biplanes the permissible limits are 0.9 to 1.2. We shall assume a value of 1.0 as a good conservative figure.

Under these circumstances, we need only correct our maximum K_y as for an orthogonal biplane with gap/chord ratio of 1. The correcting figure for this as given in Part 1, Chapter 8 is 0.81, but Dr. Hunsaker's experiments have shown that at maximum lift a better factor of 0.86 may be employed.

Since at high angles the tail surfaces will also be providing some lift, we may safely use this figure. Any obstruction between the wings, such as the body, will diminish the K_y, and in a twin-engined machine this effect would become quite serious, but it need not be considered for a single-engine type. Constructors, in fact, using a correcting factor of 0.81 or 0.82 and neglecting the lifting effect of the tail surfaces have found their landing speeds surprisingly low.

The maximum K_y for the R.A.F.6 is 0.00310. Extending the equation $W = K_y A V^2$ to include the correcting factor of 0.86, we have when $V = 43$ m.p.h.

$$1850 = (0.0031)(0.86)A(43)^2$$

$A = \dfrac{1850}{(0.0031)(0.86)43^2} = 376$ sq. ft., which is taken to include the ailerons.

lateral stability. For longitudinal stability to place the stabilizer at 3 deg. to the wings is a good setting prior to a wind tunnel test.

Position of Center of Gravity

In order to fix the position of the center of gravity, a vector diagram for the whole machine is necessary. But to draw a vector diagram, a wind tunnel model test is necessary, and in the model we have already fixed the positions of the various parts of the machine. To draw a probable vector diagram

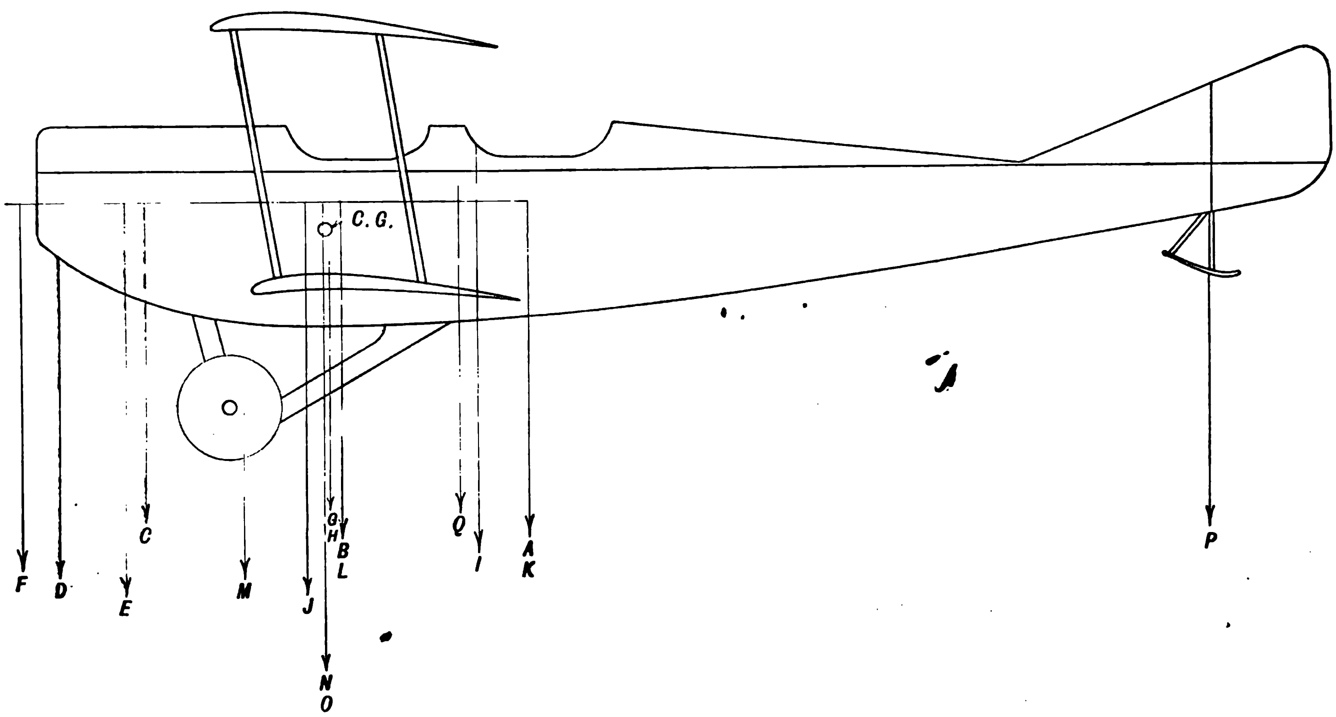

FIG. 1

without a model test is a most difficult matter. We have experiments to show the vectors for an orthogonal biplane alone, but with every different tail setting, body, shape and landing gear, we have a different vector diagram.

The best that can be done in preliminary design is, therefore, to make as shrewd a guess as possible, and to draw comparisons from as many model tests as possible. We have fortunately at our disposal the results of the tests on the Curtiss JN-2, which is almost identical with the design we are following. The vector diagram of this machine is shown in Fig. 2. Eiffel 36 wings are used, but in general arrangement

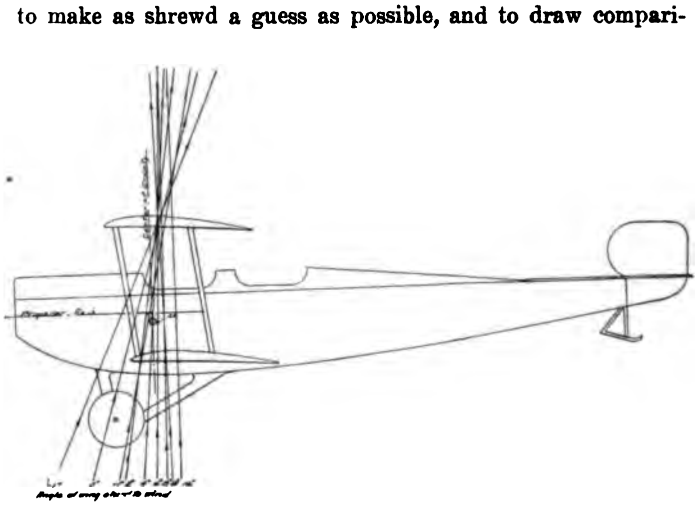

FIG. 2

The questions of aspect ratio is again only partially dependent on efficiency, and a large aspect ratio introduces structural difficulties and tends to lateral instability. We shall assume an aspect ratio of 7 to 1 on both planes. If the planes are slightly raked, the aspect ratio will be 7 for the mid-line of the wings.

If x = chord in feet, $7x$ = span, and we have $14x^2 = 376$ and chord = 5 ft. 2 in. and span = 36 ft. 5 in.

We shall now fix purely on empirical grounds the length of the machine and the size of the control and fixed surfaces.

The designer is always tempted to shorten the length of the machine and to rely on a large stabilizer placed at a big negative angle, to secure static longitudinal stability. But in dynamical stability, it cannot be too strongly emphasized that damping is also essential, and damping improves rapidly with the length of the stabilizing arm. Too short a length would give rapid, undamped oscillations. The overall length of the Curtiss JN-4, 27 ft. 3 in., is the outcome of several years' practical experience, and is probably a most suitable figure.

By empirical rules, such as outlined in Part 2, Chapter I, the control surfaces may be fixed approximately at

Ailerons	35 sq. ft.	
Horizontal stabilizer	28 sq. ft.	(at an angle of 3 deg. to the wings)
Elevator	22.0 sq. ft.	
Rudder	12.0 sq. ft.	
Vertical fin	4 sq. ft.	

The question of lateral stability is one which still requires much investigation. Purely on empirical grounds, we can say that with no sweepback, but a fin of the above size with a dihedral between the wings of 2 deg. will secure a moderate the machine is almost identical with ours. In Fig. 1 is shown a side view of this machine, with the vector diagram of a wind tunnel test.

The center of gravity is indicated in the sketch, and lies on the 4 deg. vector. Such an arrangement will give an adequate amount of longitudinal static stability. The propeller thrust passes through the center of gravity and does not, therefore, affect the stability in normal flight. Let us suppose that 4 deg. is the normal angle in flight, as it very probably would be. Then if the machine dives to 2 deg., the 2 deg. vector passing in front of the center of gravity will tend to pitch the airplane back to 4 deg. If the machine goes to a higher angle, say 8 deg., the vector will be behind the center of gravity and will give a counter clockwise moment about the center of gravity tending to restore it to 4 deg. again.

The next step is to see whether we can balance our machine about this point both in a vertical and in a horizontal plane. Before drawing up our three general arrangement views, we must go into a number of points connected with chassis design, but we can use the side view of Fig. 1 with slight modifications for our balancing up.

We shall employ the usual method for finding the center of gravity of a system consisting of a number of small bodies. That is, we choose a plane somewhere in the system as an axis and take moments about it, finally dividing the sum of the moments by the total weight of the members to find the equivalent moment arm, or distance of the center of gravity from the axial plane. In this case we shall take the axis at the rear propeller flange, that being the usual practice.

The following table illustrates the method in detail. The designations refer to Fig. 1, where the positions of the centers of gravity of the various elements are indicated.

Designation	Name of Element	Weight, lb.	Dist. from propeller flange (ft.)	Moment about propeller flange (ft. lbs.)
A	Pilot	165	10.57	1745
B	Passenger	165	6.67	1100
C	Engine	360	2.59	932
D	Radiator	50	0.77	39
E	Water	40	2.36	94
F	Propeller	35	—0.17	—6
G	Gasoline tank	30	6.44	193
H	Gasoline and oil	220	6.44	1417
I	Rear instrument board	20	9.46	189
J	Forward instrument board	20	5.90	118
K	Rear control	12	10.57	127
L	Forward control	12	6.67	80
M	Chassis	74.5	4.59	342
N	Wings	261	6.32	1649
O	Interplane bracing	91.5	6.32	578
P	Tail	51	24.75	1262
Q	Body	246	9.11	2241
	Totals	1853		12,100

Dividing the moment by the total weight of the machine, we see that the center of gravity of the machine is 6.53 ft. back of the moment axis. This brings it to a position virtually coincident with the tanks, and about one-third of the chord from the leading edges of the wings, which is virtually the position chosen from the vector diagram. If the center of gravity does not come to the desired position at the first trial, it may be forced to do so by manipulation of the weights, shifting the engines, pilot, etc., slightly forward or backward, as the need may be.

A similar method, with moments taken about the ground line, is used to give the vertical position of the center of gravity. The actual work for this computation is omitted in order to avoid confusing the figure.

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Barnwell's "Airplane Design."
"Experimental Analysis of Inherent Longitudinal Stability for a Typical Biplane."
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Chapter VIII

General Principles of Chassis Design

The design of landing gears is among the most complex of the problems which confront the aeronautical engineer, due to the many conflicting factors which must be taken into consideration and. so far as possible, reconciled.

General Proportions

The height of the chassis is dictated by the necessity of providing ground clearance for the propeller and by the angle of attack which is desired in starting and in pulling up after touching the ground. The track must be sufficient to insure against overturning when making a landing on rough ground, yet not so great that the striking of a soft spot by one wheel will give rise to an excessive moment tending to spin the machine around. The fore-and-aft location of the wheels is determined by the requirements of longitudinal stability on landing. The structure must be strong enough to withstand side thrusts and twisting moments due to alighting on one wheel, as well as the large direct dynamic stresses which are set up when an airplane lands without sufficient flattening out of the angle of descent. Lastly, the means of shock absorption must be of such quality and number that they will permit of high speed along the ground and of heavy landings without breakage of the shock absorption means itself and without danger of the "bottoming" of the axle in its guides. The play of the absorbers should also be large enough so that the dynamic landing and taxying loads previously alluded to will not reach excessive values. Each of these conditions will now be taken up in turn, and discussed in detail.

Chassis Height

Ordinarily, the most important factor here is the protection of the propeller. With a conventional shock absorber, allowing a total displacement of five or six inches, the propeller clearance with the machine stationary should be not less than twelve inches, thereby insuring a minimum clearance of six inches when the shock absorber has its maximum displacement.

Under some circumstances, the governing condition may be the angle of incidence to which it is desired to pitch the machine. The greatest possible angle should, in general, be at least as great as that which corresponds to the burble point, and it may be advisable, if a very short run after landing is required and brakes are not desired, to make it somewhat larger than this. The following example will illustrate the use of this condition:

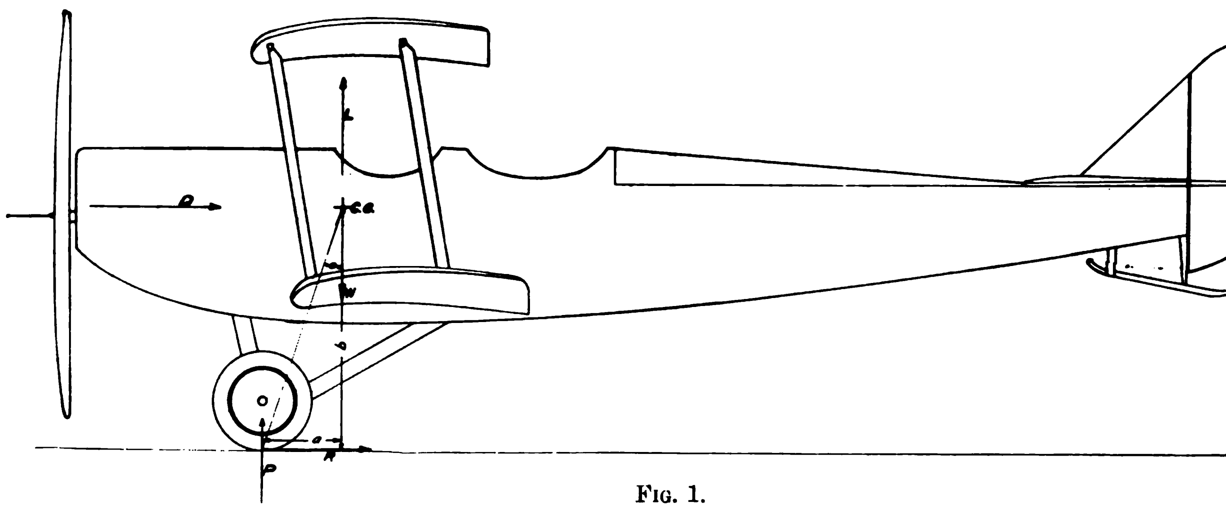


Fig. 1.

A two-seat tractor biplane is 26 feet long, and the horizontal distance between the axle and the point of contact of the tail skid is 20 feet. In normal flight, when the line of thrust is horizontal, the wings are set at an angle of incidence of 4 deg., and the point of contact of the tail skid is 2 ft. vertically below the line of thrust. It is desired to find the least distance of the lower rim of the tire below the thrust line which will permit the assumption of an angle of 18 deg. Since the wings are at an angle of 4 deg. in normal flight, the maximum angle between the thrust line and the ground must be 18 deg.—4 deg., or 14 deg. Tan 14 deg. = .249, and the difference in heights of the wheels and tail skid is therefore $20 \times .249$, or 4.98 ft. It is then evident that the required height is $4.98 + 2$, or virtually 7 ft., a much greater height than would be necessitated by propeller clearance alone. If such large angles must be attained it is worth while to sacrifice something from the perfect symmetry of the body by putting most of the longitudinal curvature on the lower surface of the body, thereby bringing the tail skid more nearly into the line of thrust and decreasing the height, and consequently the weight and resistance, of the body, to an extent which more than compensates for any loss of aerodynamic efficiency of the body.

The tread, or track, is, on airplanes of conventional size, from 5 to 7 ft., except on slow pusher biplanes with long skids, where it may reach values as high as 13 ft.

Location of Chassis with Respect to C. G.

In Fig. 1 are shown the forces which come into play while the machine is running along the ground. It is evident that, if we assume the thrust line to pass through the c. g., and the machine to be in equilibrium with the elevator either in neutral or diving position, so that the resultant air pressure with the body horizontal passes through or slightly behind the c. g., the moment $P \times a$, due to the upward reaction of the ground, must be at least equal to the moment $R \times b$, due to the tractive resistance. We then have, since we may write

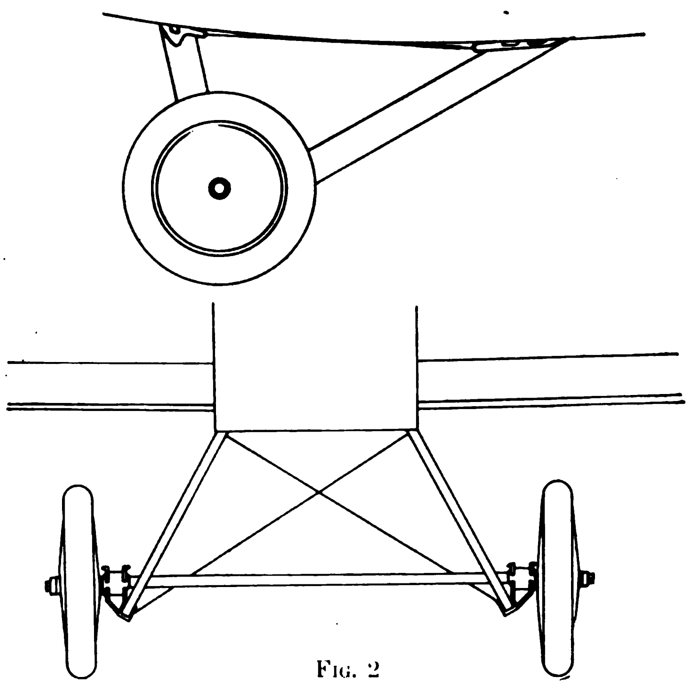


FIG. 2

$R = \gamma P$, $P \times a = > \gamma P \times b$, or $\frac{a}{b} = \tan\theta = \gamma$, where γ is the coefficient of tractive resistance. If we assume $\gamma = 1/10$, which is perhaps a fair value for a vehicle with large and flabby pneumatic tires running over smooth grass, we have $\theta = 5$ deg. 44 min. Since, however, it is necessary to allow for soft ground, where the coefficient of tractive resistance will be much increased, as well as for ruts and changes of slope, which introduce a backward component in P itself, it is obvious that θ will have to be considerably larger than the value given above. Lieut. Col. B. Q. Jones, U. S. A., states that the best practise indicates a value of 13 deg. 10 min. for θ.

Stresses and Structural Considerations

An unequal distribution of stress between the wheels, due to landing on one wheel before the other or to a difference in ground conditions between the two tracks, produces a moment tending to twist the axle about a vertical axis. This moment is carried to the struts or skids by means of axle guides or radius rods, and thence to the body. A landing on one wheel generally involves a sideways motion, and the resulting sideways blow is usually carried by the cross-wires, although some landing gears, particularly on speed scouts, have inclined struts which, acting either in tension or compression, take the place of the wires in this respect. In order to resist side blows, too, a special wheel construction is necessary, as an ordinary wire wheel, such as is used on motorcycles, will be completely wrecked by a relatively small side blow against the rim. It may be laid down as a rule that the length of hub should be at least twice the diameter of the tire, and three times is preferable. The obliquity of the spokes is thus greatly increased.

Direct dynamic loads in landing have already been discussed in this course (see Part II, Chapter 6).

Shock Absorbers

Rubber and steel springs are the only substances which have been widely used as shock absorbers. Of these, rubber has proved by far the more satisfactory, due to its easy fabrication and replacement, its greater energy-storing capacity (500 to 1,000 ft. lbs. per lb., as against 10 to 20 ft. lbs. per lb. for steel), and, most of all, the fact that it actually absorbs and dissipates the energy, instead of merely storing it and giving it out again. If steel springs are used, some auxiliary device must be employed to dissipate the energy in the form of heat. In the case of leaf springs, this is done in a fairly satisfactory degree by friction between the leaves, and such springs have been used on light machines built for smooth fields, but they do not afford sufficient give for heavy work. Where helical springs are used, as on many large pusher biplanes with four-wheeled landing gears, they are usually in combination with a hydraulic or pneumatic shock absorber. As an illustrative problem, we shall give the complete calculation of a rubber shock absorber for a two-seater tractor biplane weighing 2000 lbs., having a gliding angle of 1 in 7, and a speed range of from 45 to 90 miles an hour. We shall start with the assumption that the heaviest landing shock which needs to be provided against is that due to landing at 45 m.p.h. on a slope of 1 in 6 without any attempt at flattening out.

The shock absorbers will be made up of rubber rings 2 in. in diameter, and 2 in $\times \frac{5}{16}$ in. in section. We shall use a two-wheeled landing gear, so that each wheel will have an initial load of 1000 lbs. The initial stretch will then be $\frac{1000\ l.}{E \times A \times n}$. E, the modulus of elasticity, may be taken equal to 300 lbs. per sq. in. for rubber of good quality. A, the total cross-section area of one ring, is $2 \times 2 \times \frac{5}{16}$, or $1\frac{1}{4}$ sq. in. l, the length, may be taken equal to one-half the perimeter of the rings, or $l = \frac{2 \times \pi}{2} = 3.14$ ins. n is the number of rings employed.

We then have the initial stretch $= S = \frac{1000 \times 3.14}{300 \times 1\frac{1}{4} \times n} = \frac{8.39}{n}$, and s, the deflection under any load, $= \frac{16.78F}{Wn}$, F being the total load.

On landing on a slope of 1 in 6 at a speed of 66 ft. per sec., the vertical speed is 11 ft. per sec., and the kinetic energy is $\frac{(11)^2 \times W}{2g}$, or $1.88W$ ft. lbs. The potential energy possessed on touching the ground is $\frac{WS}{12}$, making a total of $W\left(1.88 + \frac{s}{12}\right)$, or $W\left(.94 + \frac{s}{24}\right)$ on each wheel.

The energy absorbed by the shock absorber is $\frac{F}{2} \times \frac{s}{12} = \frac{Wns^2}{402.7}$. Equating this to the energy possesed by the machine, $\frac{Wns^2}{402.7} = W\left(.94 + \frac{s}{24}\right)$. We shall assume a deflection of

5 in. Then $n = \dfrac{402.7\,(.94 + 5/24)}{25} = 18.54$. We shall use 18 rings and recalculate. Then $\dfrac{18\,s^2}{402.7} = .94 + \dfrac{s}{24}$, and $18\,s^2 = 378 + 16.78\,s$. $s = 5.05$ in.

The stress in the rubber is $\dfrac{5.05}{\pi} \times 300 = 482$ lb. per sq. in., as against a breaking strength of 800 lbs. per sq. in., and the load on the chassis $= \dfrac{2W\left(1.88 + \frac{s}{12}\right)}{\frac{s}{12}} = \dfrac{4000 \times 2.30}{.42} =$ 21,900 lbs. At the instant of greatest displacement, then,

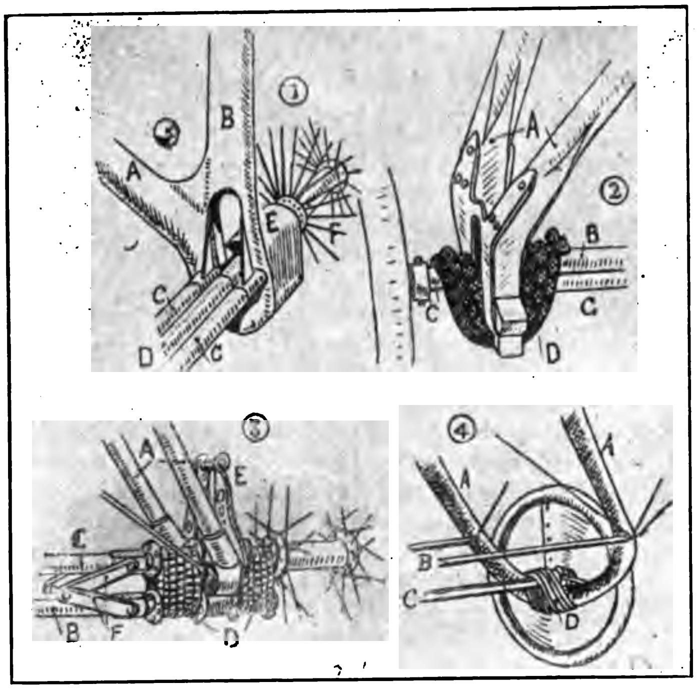


FIG. 3. ARRANGEMENTS OF RUBBER CORD AND RINGS USED ON VARIOUS FRENCH MONOPLANES.

every part of the machine has on it a downward load of eleven times its own weight.

We have so far assumed the most unfavorable condition with respect to the tires: that is, that they do not deflect at all. We shall now work the same problem on the assumption of the most favorable conditions reasonably to be expected, that 26 × 4 tires are used, and that they are pumped to such pressure that they collapse 2 in. under the same force which causes the shock absorbers to yield 5 in. The effect of this is equivalent to increasing the length of the shock absorbers by 40 per cent. We then have $F = \dfrac{Wns}{23.49}$, and $\dfrac{Wns^2}{563.8} = W\left(.94 + \dfrac{s}{24}\right)$. s now equals 7 in., and n is therefore $\dfrac{563.8\left(.94 + \frac{7}{24}\right)}{49} = 14.28$. Only 14 rings will be needed under this assumption.

Actually, a 2500-lb. airplane usually employs about 12 rings on each bridge, and is therefore unable to sustain conditions as severe as those which we have assumed. In order to provide the desired shock absorption capacity, machines of the size which we are considering use 26 × 4 or 26 × 5 wheels. These wheels weigh, complete with tire, from 17 to 25 lbs. each, and the manufacturers recommend that they be pumped to 60 lb. pressure.

In place of rubber rings, as specified above, rubber cord woven from many strands may be used. Some recent tests on 5/8 in. cord, made up of 180 strands $\frac{1}{32}$ in. square, showed a modulus of elasticity of 434 lb. per sq. in. The breaking load was not determined. During these tests, it was discovered that of the "permanent" set which rubber cord takes after being stretched, 40 per cent of its original length almost

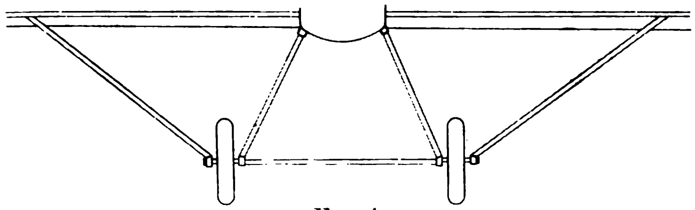


FIG. 4

entirely disappears after 17 hours of rest, the rubber regaining all its original properties.

Types of Chassis

We may, in general, divide chassis into three classes: self-contained chassis with wheels alone, chassis with two principal wheels and a tail skid, and chassis built up around one or two long skids as a basis. There are also numerous compromise designs, which it is difficult to assign to any one class. Nearly all the land machines now built in the United States come into the second of these classes, although all three types had some vogue here at one time.

Chassis with wheels alone were first used by Curtiss. Due

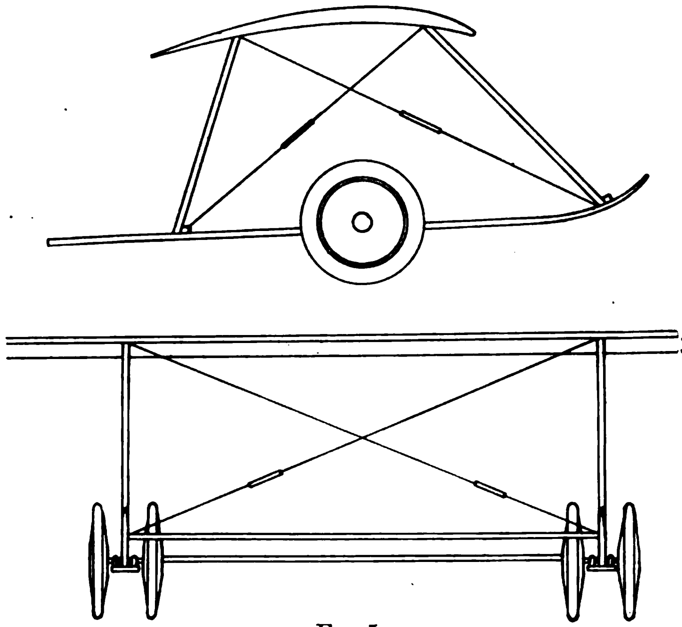


FIG. 5

to their complexity and considerable weight and resistance, they are now seldom employed except on heavy machines, particularly pusher biplanes designed for gun-carrying, where they are generally combined with helical steel springs and hydraulic shock absorbers. Such a chassis may have either three or four wheels, four being the more common. The fore-and-aft distance between the two pairs of wheels must be considerable, in order to insure against the machines falling over on its tail as it is brought to rest.

Chassis with two principal wheels and a tail skid, either with or without one or two subsidiary, and usually unsprung, wheels in front, are nearly universally employed on machines of medium and light weight, except on the very slow and lightly-loaded pusher biplanes. The framework of such a landing gear is reduced to its lowest terms, consisting, in its conventional form, merely of two Vs, closed at the top by the lower body longerons, and separated at the top by a distance equal to the width of the body, and at their lower vertices by a little less than the track of the wheels. The bottoms of the Vs are connected by a strut. The two wheels are mounted just outside the Vs, either on a single axle or on separate axles hinged at the center, usually the former. The present practice is to incline the axle guides so that the wheels travel backward somewhat as they rise in the guide slots, but this causes excessive wear on the front of the slots, and experiments at the Signal Corps school indicate that no harmful results follow the elimination of the backward slope. Fig. 2 gives a diagrammatic view of a typical chassis of this class.

On speed scouts, where the reduction of resistance is of the utmost importance, the chassis is sometimes so built that it forms a unit with the wings. This makes possible the elimination of all wiring from the chassis, and in a few cases from the wing panel as well. Fig. 4 represents a machine so braced.

FIG. 6. SOPWITH CHASSIS, SHOWING COMBINATION OF TWO WHEELS WITH SHORT SKIDS

Chassis based on long skids are used on Farman, Caudron, and Wright biplanes. They generally embody four comparatively small and light wheels arranged in a straight line, one pair of wheels, about 18 inches apart, being attached to each skid. Each pair of wheels has its own axle, and radius rods are used to prevent the axle twisting with respect to the skid. Rubber shock absorbers are used. In some cases the skids are carried up in front to a forward elevator, as in the Maurice Farman, or in the rear to the tail, as in the Caudron. Such a chassis is illustrated in Fig. 5.

Among the hybrid types, involving some of the features of all three classes, one of the most interesting is the old Nieuport. In this there were 2 pairs of struts, each pair forming a V with the vertex downward. At the bottom of these was mounted a comparatively short skid. Below the skid a leaf spring, of semi-elliptic form, was clamped, and the wheels were at the extremities of this spring. The rear of the skid acted as a tail skid, and, in addition, made a very effective brake, as it was close to the c. g., and consequently carried a considerable portion of the weight.

Brakes and Braking

All the devices which have been brought forward for checking the speed of airplanes after touching the ground fall naturally into one of two divisions: depending either on air resistance or ground resistance. The best example of a brake depending on air resistance is the airplane itself. We have already mentioned the desirability of being able to depress the tail to such a degree as to secure a very large angle of incidence, not only so that we may land at low speed, but also so that the drag may be increased, and thus aid in checking the speed after landing.

It has often been proposed that air brakes, consisting of surfaces normally lying parallel to the line of flight, but capable of being pulled around approximately normal to that line, should be provided. If two such surfaces, one on either side of the body and at a considerable distance from the longitudinal axis, are furnished, one at a time may be pulled out to act as a drag and assist in turning the machine in a small circle while taxi-ing, or both may be used at once as a brake. The trouble with all air brakes is that they rapidly lose their effectiveness as the airplane begins to slow up. They have, on the other hand, the advantage that their force is exerted well above the c. g., so that there is no tendency to stand the machine on its nose.

Brakes depending either directly or indirectly on friction, with the ground for their retarding power, may be subdivided into wheel brakes and sprag or claw brakes. Wheel brakes, although they can be made very powerful, are hardly ever used, due to their danger, which lies in the difficulty in releasing them quickly, and in the fact that they have no tendency to release themselves automatically as the machine starts to pitch over on its nose.

Claw brakes are more used than any other type. They are usually attached to the strut, which lies just below the axle in a V-type landing gear, and are hinged to that strut. The claw on the end can be brought to bear against the ground by a lever within reach of the pilot. The advantage of such a brake is that, being in back of the forward point of suspension, the claw tends to release itself as the machine starts to dive, pivoting about the point of contact of the wheels with the ground. From this point of view, the brake should be as far back as possible, but the available retarding force is greater and the construction is simpler when it is kept near the chassis proper. The tail skid itself acts as a very efficient claw brake if it is so arranged as to carry a considerable portion of the load. If extra quick stops are desired from a machine, whatever type of brake is used, the wheels should be placed farther forward of the c. g. than usual, thus permitting a larger moment to be applied at the ground level without upsetting the machine, making it easier to get the tail skid down in contact with the ground immediately after landing, and throwing a larger portion of the weight on the tail skid or sprag brake, if one is used.

It is of interest to determine the retarding force required to bring a machine to rest in a given distance. If, for example, we wish to land a 2500-lb. machine at 45 m.p.h. and bring it to rest in 200 ft. after touching the ground, we have $s = \frac{v^2}{2a}$, or $a = \frac{(66)^2}{600} = 7.26$ ft. per sec. where a is the necessary deceleration. F, the average retarding force, $= \frac{W \times a}{g} =$ 565 lb., a force which might easily be secured without the use of any brake save that afforded by the wings themselves.

NOTE—The vector diagram for the JN-2, to which reference was made in discussing the design of a secondary training airplane in the last installment of the course, is reproduced herewith. The weights are omitted in order not to confuse the diagram, but the position of the c. g. is that indicated previously.

References for Part II, Chapter 8

G. de Havilland in *Flight*, March 9, 1912.
F. W. Lanchester in *Engineering*, May 22, 1914.
"The Flying Machine from an Engineering Standpoint," by F. W. Lanchester; p. 80.
"Relative Positions of Propeller Axis, Center of Gravity and Wheels," by Capt. B. Q. Jones; AVIATION AND AERONAUTICAL ENGINEERING, Nov. 1, 1916.
"L'Essor et l'Atterissage," by Maurice Percheron.
"La Construction des Organes de Contact avec le Sol," by P. James; *Revue Générale de l'Aéronautique*, Jan., 1914.
"Notes on Aeroplane Shock Absorbers of Rubber," by J. C. Hunsaker; AVIATION AND AERONAUTICAL ENGINEERING, Sept. 1, 1916.

Chapter IX

Type Sketches of Secondary Training Machine General Principles of Body Design

In Fig. 1 are shown three views of a secondary training machine, very similar to the JN-2, and in accordance with our figures of Chapter 7.

A few modifications have been made in the process of drawing up the machine from the figures given in Chapter 7. The figures there were derived from empirical formulas, but in the present stage of the art it cannot be too strongly insisted that no empirical formulas hold with absolute rigidity, and that "eyeability" is almost as important—except in the case of the stabilizer and elevator, on which more data is available. Thus the rudder has been reduced in area from 12 to 10 sq. ft., and the vertical fin from 4 to 3.5 sq. ft.

The stabilizer and elevator have been left unchanged. In drawing the plan view of the machine, modifications were also found necessary in the ailerons. The original scheme was to place the ailerons on the top plane only. But in order to secure the necessary area it was necessary, with the spar position selected, to make the ailerons very long and bring them in comparatively close to the body (with an overhang on the top plane this difficulty would not have occurred), and ailerons brought in close to the body have an insufficient leverage for part of their surface. The better plan seemed to be, therefore, to place the ailerons on both surfaces. Their area was also slightly increased, from 38 to 42 sq. ft. total area.

It must be insisted upon again that this machine is not a perfect specimen of its type. For instance, had an overhang been employed as on the JN-4, the aileron area of 35 sq. ft., with its greater lever arm, would have been amply sufficient. Also the outer strut would have been almost at the mid point of the aileron, thus permitting the use of a single aileron post; whereas in the present case we are obliged to use two aileron posts.

Another poor point is that the tail skid abuts directly on the rudder post. The control surfaces should never be so placed as to sustain injury by an abrupt landing, as might be the case in this arrangement.

A drag wire is shown carried from the top of the inner strut to the engine. This helps to keep the body from twisting under the effect of gyroscopic forces on the engine, and also to relieve the drag bracing. Nevertheless, in computing the drag bracing the effects of such a wire are totally neglected.

General Requirements in Body Design

These may be very briefly summarized:

(1) Stream-Line Form

The power plant and personnel must be enclosed in a form approximately stream-lined. The general shape of the body is largely determined by the size and shape of the engine selected. For the vertical six-cylinder engine the body may be narrow and deep. For a V cylinder engine, a wider but shallower body is advisable, and with a rotary engine a body of very large maximum diameter. But consistent with structural and other considerations, a body should be selected which gives minimum aerodynamic resistance. The best form of body would, of course, be symmetrical about an axis. Some data for the resistance of airplane bodies has been given in the first part of the Course, but there is no doubt that considerable improvement is possible in this direction, possibly by employment of monocoque construction. Where a four girder body is used, and attempts are made to secure streamline form, the designer must guard against excess weight.

(2) Fin Area of Body

A flat bottomed body may be very helpful in securing longitudinal dynamic stability. A body with flat sides has to be handled carefully. It is equivalent to a long fin, with most of the fin area aft of the center of gravity, and this tends to head a machine into the wind—an advantage if the effect is not excessive. Such fin area is, however, best secured by the use of a vertical fixed fin. With a large flat sided body, it is as well to investigate yawing moments in the wind tunnel. One of the reasons why totally enclosed bodies have not come into use is that with their large fin areas, they have a tendency to spinning.

(3) Length of Body

Apart from the necessary length of body to give sufficient arm to the tail surfaces, it is important that the tail surfaces should be far enough away from the wing so that the wash of the wings should not affect them too much.

(4) Provision for Pilot and Passenger

The necessary requirements are obvious. To protect the face of the passenger, a transparent lip is generally fitted on the front edge to deflect the air upwards. The back of the pilot's head may be stream-lined with a suitable projection. Specification 1002 gives standard arrangements for pilot's and passenger's seats.

(5) Engine Installation

Should be readily accessible and cowling easily removable.

(6) Gasoline Tanks

Should be near the center of gravity of the whole machine, so as to disturb balance as little as possible as fuel is consumed. Where it is impossible to place the fuel supply directly over the center of gravity, the gasoline and oil may be made to balance one another approximately.

(7) Engine Foundation

Must be rugged to prevent loosening up of the bolts by vibration, transmission of the torque of the engine to the body,

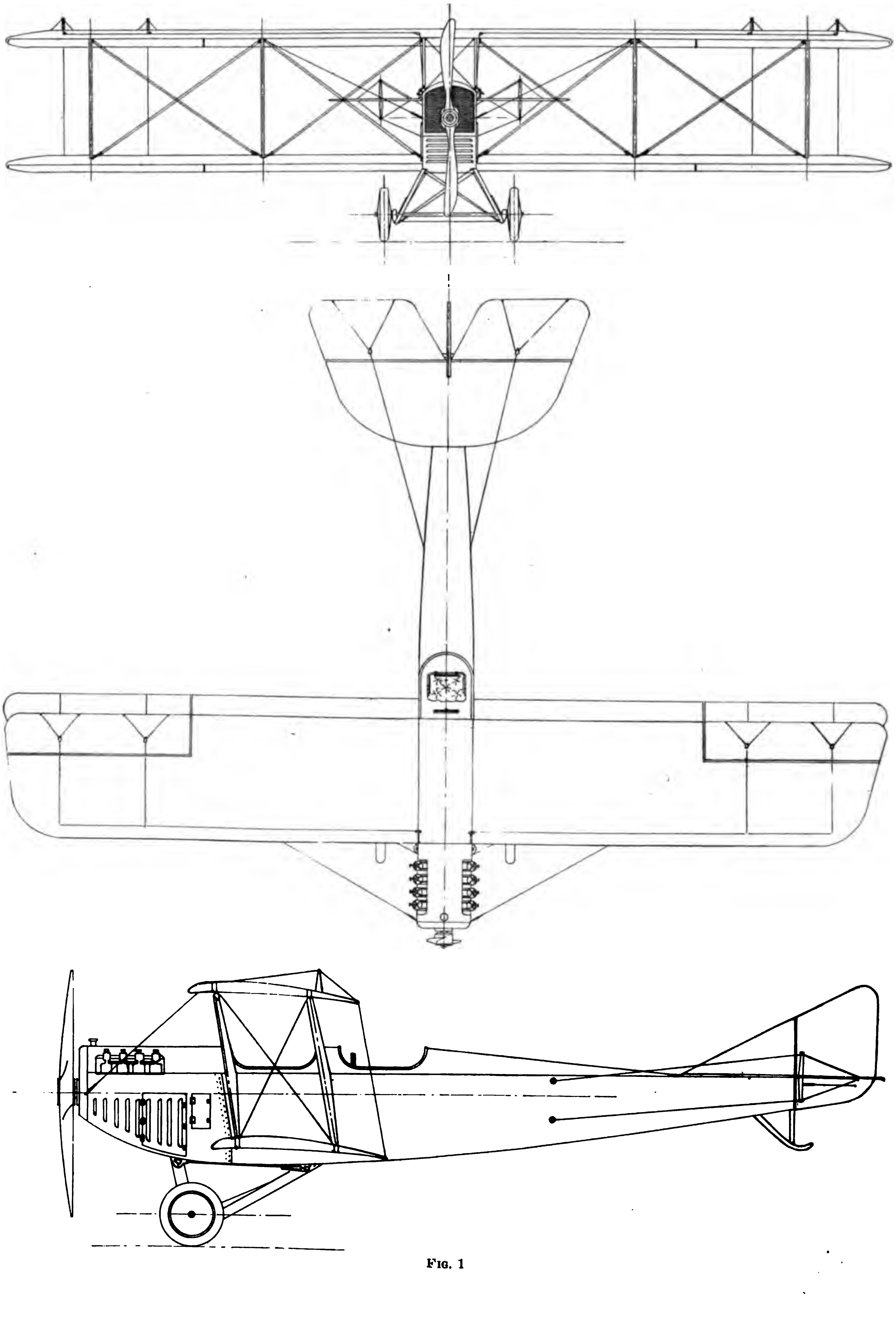


Fig. 1

and breaking loose in a bad landing. Nevertheless, the foundation should be flexible enough so that slight engine vibration is easily taken up. The following example will illustrate the forces on the foundation bolts due to engine torque: Six cylinder 120 h.p. 1200 r.p.m. Torque $= \frac{120 \times 550}{2\pi \times 20} = 525$, then if F is force on either side, $2F \times .575 = 525$, where .575 is half the distance between engine bed bolts, and the force on either side is 457 lb.

(8) Engine Must Be Secured Against Weaving

When the airplane pitches, there is a tendency owing to gyroscopic action of the propeller, for the engine to "weave" either to right or left. Diagonal members in the plane of the engine bearers as well as wires are often used. The ideal engine foundation would seem to be of pyramidal form.

(9) Strength of Body

The body must be strong enough to withstand (a) air loads due to tail surfaces, (b) dynamic loads in the air, (c) loads on landing.

These are but a few of the requirements in body design. Numberless points arise in detail work, in which experience and care, and not general rules, are necessary.

Formulas for Spruce Compression Members

The most reliable data on spruce struts—the material in which we are most interested—is given in Dr. Hunsaker's note to which reference is appended. Experiments were carried out on Maine white spruce, West Virginia white spruce, and Oregon red spruce. Values varied so much for each specimen that it would be unsafe to use them definitely for wood of varied origin, of varied position in the log, and degree of seasoning, and in actual construction tests on specimens are always necessary.

In these experiments, the modulus of elasticity found by observing deflection under loading was found to be 1,825,000 pounds per square inch. Two formulas for crippling stress, defined as the crippling load divided by the area of cross section in square inches, were deduced.

(1) For long struts, $\frac{L}{K} > 70$, $\frac{P}{A} = \frac{8.72\,E}{\left(\frac{L}{K}\right)^2}$ where $P =$ crippling load $A =$ area in square inches $L =$ length in inches $K =$ least radius of gyration in inches $E =$ modulus of elasticity in pounds per square inch. (Some designers employ the ordinary Euler's formula, $\frac{P}{A} = \frac{\pi^2 E}{\left(\frac{L}{K}\right)^2}$ using a value of $E = 1{,}600{,}000$.)

(2) For short struts, $\frac{L}{K} < 70$, $\frac{P}{A} = 6500 - 46.5\frac{L}{K}$. (Some designers employ a modification of Rankine's formula: $P = \frac{A\,fe}{1+\phi\left(\frac{L}{K}\right)^2}$ where $fe = 8{,}000$ lbs. for spruce, $E = 1{,}600{,}000$. $\phi = \frac{fe}{\pi^2 E}$.)

By careful selection of spruce the crippling loads given by the above formulas can be easily secured. It was formerly customary to use a material factor of safety of 2 for the wing struts, and 1½ for body struts.

There arises a further difficulty in connection with the above formulas, in determining whether a strut is fixed or hinged at the ends. It is usually assumed that

(1) wing struts with pin joint fastenings are hinged at either end.
(2) wing struts with socket fastenings of usual type are considered as being fixed at one end, and round at the other.
(3) body longitudinals, continuous over joints, are taken as fixed at ends.
(4) body horizontal and upright struts are taken as fixed at one end and hinged at the other.

For a strut fixed at one end, and hinged at the other, the equivalent length becomes $\frac{L}{\sqrt{2}}$, for a strut fixed at both ends, the equivalent length becomes $\frac{L}{2}$. Thus the above formulas become:

(1) Ends hinged	$\frac{P}{A} = \frac{8.72\,E}{\left(\frac{L}{K}\right)^2}$	$\frac{P}{A} = 6500 - 46.5\,\frac{L}{K}$
(2) One end fixed one hinged	$\frac{P}{A} = \frac{8.72\,E}{\frac{1}{2}\left(\frac{L}{K}\right)^2}$	$\frac{P}{A} = 6500 \quad \frac{46.5\,L}{\sqrt{2}K}$
(3) Both ends fixed	$\frac{P}{A} = \frac{8.72\,E}{\frac{1}{4}\left(\frac{L}{K}\right)^2}$	$\frac{P}{A} = 6500 - \frac{46.5\,L}{2\,K}$

Body Stress Diagrams

Body stress diagrams are still on a somewhat unsatisfactory basis, and a number of different methods are adopted. Although the longerons of a body are continuous, and the cross bracing members more or less fixed, stress diagrams are always drawn as if it were entirely a pin-jointed structure. Subsequently, compression members are treated as either wholly or partly fixed at the ends. This is inconsistent but probably all that can be done, without very lengthy refinements.

Factors of safety have been specified in a number of ways, of which we have noted some already.

Army Specifications 1000, 1001 and 1002

Air speed, 100 miles an hour. Angles of incidence of fixed horizontal tail surface, minus 6 deg.; elevator surface, minus 20 deg. Factor of safety not less than 2.5. This is based on the forces met with when the machine is violently righted after a rapid dive. It takes care solely of the air loads due to tail surfaces. When in the air the body is supported at the hinges of the wings, and the air loads are not transmitted to the part of the body forward of the hinge pins. It can be seen that this is by no means an ideal specification. It has also been criticized on the ground that no pilot can, under ordinary conditions, exert sufficient force to move the elevator to such a position.

Army Specification 1003

Body forward of the cockpit shall be designed for a factor of safety of ten (10) over static loading conditions with the propeller axis horizontal. Body in the rear of cockpit shall be designed to fail under loads not less than those imposed under the following conditions:

(a) Dynamic loading of 5 as the result of quick turns in pulling out of a dive; (b) superposed on the above dynamic loading shall be the load which it is possible to impose upon the elevators, computed by the following formulas: $L = .005AV^2$ where A is the total area of the stabilizing surfaces, i.e. elevators and fixed horizontal surface, and V is the horizontal high speed of the airplane. The units are kilograms, square meters, kilometers per hour; (c) superposed on this loading shall be the force in the control cables producing compression in the longerons.

This specification is sounder than the previous one. It imposes the air load on the rear part of the body, which is as it should be, and provides a sufficient dynamic loading for the forward part of the machine.

Another Suggested Method

In the author's opinion, the stress diagrams should be even more complete. They should include calculations (a) carried through on the air loads, (b) calculations carried through on the landing loads, specifying some landing speed, a gliding angle, and travel of shock absorber.

A Detailed Example of Stress Diagram

In Fig. 2, is shown the skeleton framework of a JN-2 body, which fits in with our design of a standard airplane body.

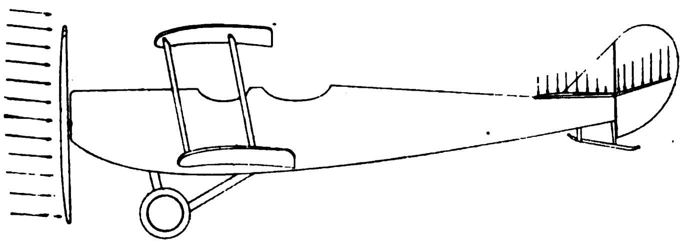


FIG. 2

In accordance with the preceding paragraph, we should draw two diagrams for it:

(1) With the body and tail surfaces in position shown in Fig. 2, horizontal tail surface at minus 6 deg., elevator surface at minus 20 deg. The air loads on these surfaces can be computed as follows: Lacking precise experimental data, we may assume that in the worst possible case, the pressure on the tail = $.0026AV^2$ area of stabilizer and elevator = 50 sq. ft. V, the highest speed attainable during a dive, may be taken as 100 m.p.h. F, the tail load, then equals $.0026 \times 50 \times (100)^2 = 1300$ lb.

With these air loads computed, a stress diagram can be easily drawn as for a simple cantilever with supports at the rear body hinges. As an article in AVIATION and AERONAUTICAL ENGINEERING for March 1, 1917, shows, the stresses obtained in this way are in certain members smaller, in other members larger, than those obtained from the landing diagram.

(2) The stress diagram on the assumption of the landing shock leaves room for much discussion. The difficulty arises primarily from the fact that it is difficult to say what the worst landing conditions before breaking are for which a machine should be designed. Also it is extremely difficult to include all the forces in play, which may include (a) lift on the wings, (b) drag on wings and body, (c) lift and drag on the tail surfaces, (d) the reaction perpendicular to the ground, (e) tractive resistance on the wheels.

Further difficulties arise from the fact that the center of the wheels does not lie under the center of gravity of the whole machine, so that if a dynamic load is applied vertically at the wheels, the weight applied at the center of gravity gives a turning moment which must be balanced in some way or another. Two methods are suggested which seem fairly reasonable, and provide a rational method of computation. In the first method, it is assumed that the machine is gliding on a path of say 1 in 7, and hits the ground nose heavy. In

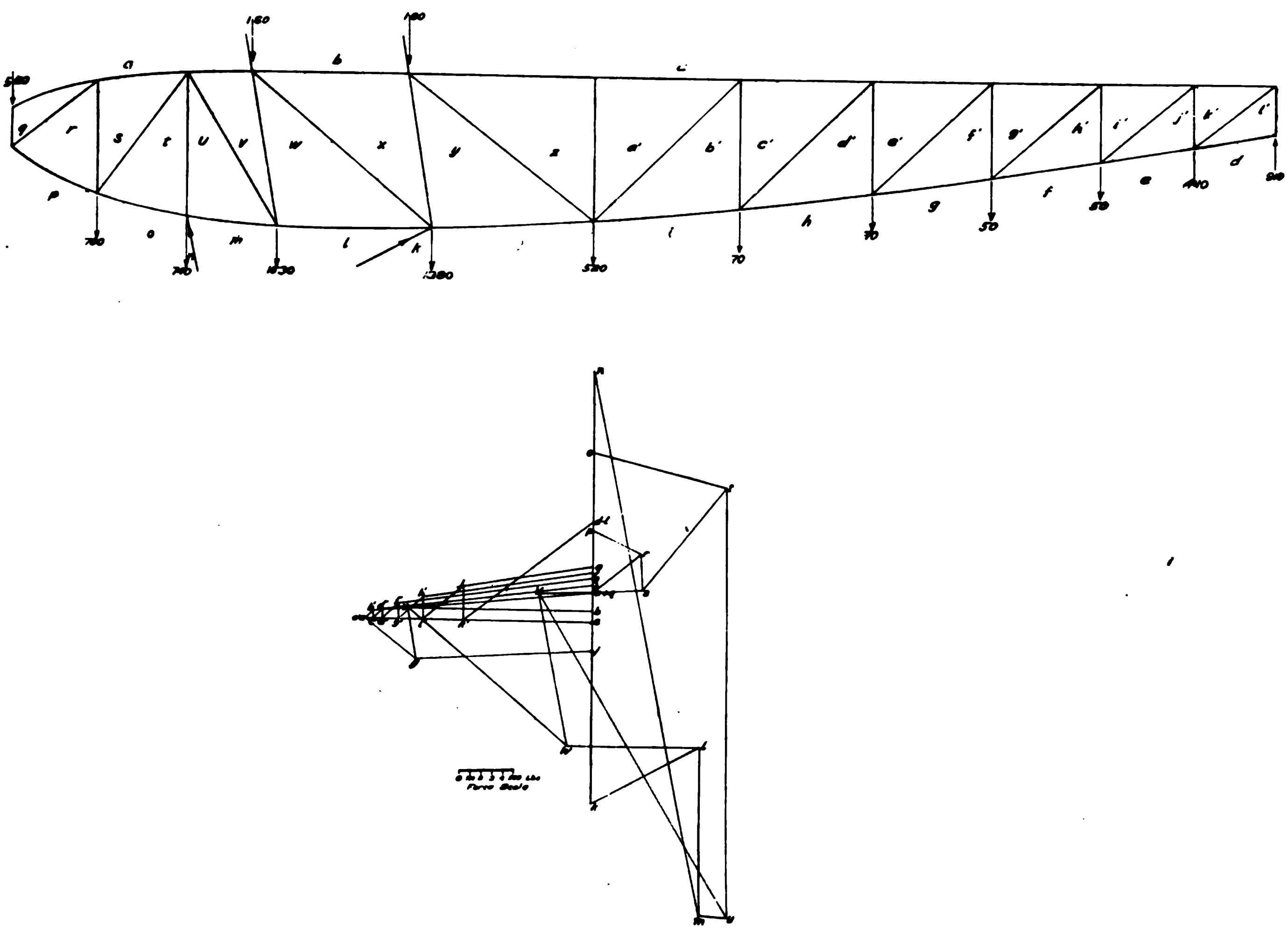


FIG. 4

such a case as can be seen from Fig. 3 the reaction of the ground may be assumed to pass through the center of gravity of the machine, and a balance of forces is obtained. The

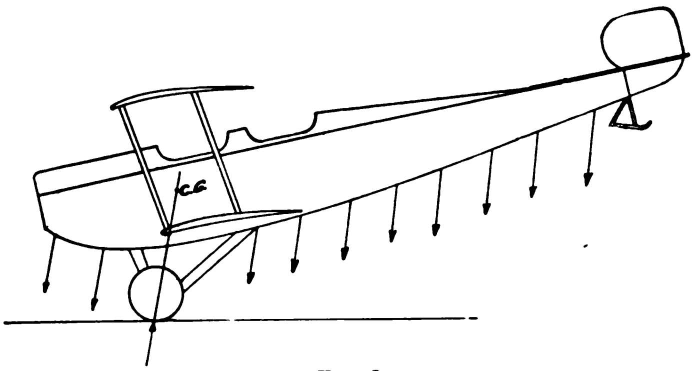


FIG. 3

dynamic load in such a case may be obtained on lines indicated in a previous section on chassis design.

In the second method, the pilot is assumed to flatten out from the glide, and then turn up to a big angle and pancake down, with the wheels and skid striking the ground simultaneously. In such a case, it is very difficult to compute the dynamic load, but a balanced system of forces is readily obtained, distributing the load between the wheels and the skids as shown in Fig. 4. The dynamic load factor there is taken as 8. In Table 1 are tabulated the stresses in various members. In the ensuing sections, dimensions will be allotted to such members.

TABLE I

BODY STRESSES

		Longerons	
aq	0	pr	490 C.
as	460 T.	ot	1250 C.
av	490 C.	mu	260 C.
bx	1680 C.	lw	1200 T.
cz	2070 C.	jy	1600 T.
ca′	2070 C.	ib′	2000 T.
cc′	1990 C.	hd′	1920 T.
ce′	1910 C.	gf′	1780 T.
cg′	1770 C.	fh′	1550 T.
ci′	1540 C.	ej′	1180 T.
ck′	1170 C.	dl′	
		Struts	
pg	560 C.	b′c′	70 C.
rs	330 C.	d′e′	90 C.
tu	3900 C.	f′g′	140 C.
vw	1410 C.	h′i′	210 C.
xy	470 C.	j′k′	[illegible]10 C.
za′	0	l′c	9[illegible] C.
		Wires	
gr	570 T.	c′d′	120 T.
st	1200 T.	e′f′	210 T.
uv	3400 T.	g′h′	300 T.
wx	1910 T.	i′j′	480 T.
yz	590 T.	k′l′	1470 T.
a′b′	100 T.		

References for Part II, Chapter 9

"Notes sur la Construction des Aéroplanes," by P. James; *Revue Generale de l'Aeronautique Militaire*, March, 1914.

"Spruce Aeroplane struts under Compression," by J. C. Hunsaker; *Aerial Ag* August 16, 1916.

Chapter X

Computation of Strength Members and General Layout of Body

In designing tension members for the body, no feature is of greater importance than the choice of terminal fastenings which will permit the development of as large a percentage as possible of the true strength of the wire or other tension member.

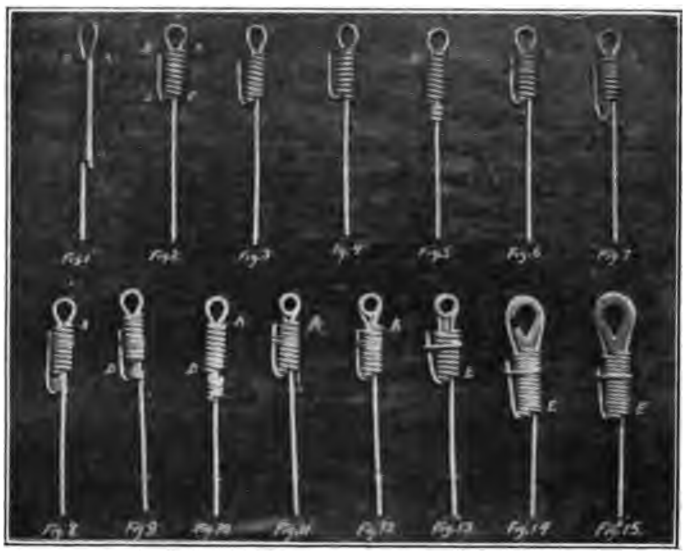
The main points to be considered in dealing with terminal connections are:

(1) The efficiency, as mentioned above.

(2) Quickness and ease of manufacture.

(3) The possibility of easy and efficient repair or replacement in the field.

(4) Reliability, i.e., the difference in efficiency between the best and poorest terminals of a series all made up in the same way should be as small as possible.



FIGS. 1-15. TERMINAL FITTINGS FOR SOLID WIRE, TESTED BY JOHN A. ROEBLING'S SONS CO.

(5) The possibilities of defects due to the use of acid and solder, overheating, imperfect bends, flattening of wire on bends, or unskillful handling of the material in the field. This requirement is obviously closely allied to that of reliability.

Extended tests on terminal connections of all types have been made by John A. Roebling's Sons Co. A summary of the most important results is given herewith, and reference to the original report is appended.

The first series of tests related to hard-drawn aviator wire. The form of terminal which was most common up to a few years ago, consisting of a ferrule made from a coil of wire, through which the wire is passed and then doubled back on itself (Figs. 2 and 3), gave very poor and uneven results, the efficiency varying from 60 to 75 per cent, with an average of 65 per cent. These efficiencies were improved by about 5 per cent when the free end of the wire, instead of being doubled back outside the ferrule, was wound three times around the standing portion of the stay.

The next type of terminal tested was similar to the last, but was dipped in solder after being made up (Fig. 1). The ferrule for such a connection may be made of a coil of wire, as previously, or of a strip of thin sheet metal, wrapped around both portions of the wire. The efficiencies obtained ran from 60 to 90 per cent, with an average of 80 per cent. These values are surprisingly low, and indicate probable damage of the wire by overheating in the process of soldering, as a connection such as this, absolutely preventing any slippage of the wire through the ferrule, should always show 100 per cent efficiency if properly made up. Tests on similar terminal fittings at the Massachusetts Institute of Technology have nearly always developed the full strength of the wire, the stay breaking near the center on every test. The soldered joints have, however, the disadvantage that they cannot readily be replaced in the field, and they are peculiarly susceptible to poor workmanship, the effects of which cannot be determined in any way until the break actually comes.

In the Roebling tests, the best results were secured by the use of tapered ferrules, winding a coil of wire into the form of a slightly flattened cone instead of a flattened cylinder, in conjunction with wedges designed to increase the friction between the stay and the ferrule as the pull increased. Such wedges may be separate members, fitted between the eye and the ferrule, in which case the wire is looped completely around to make a double eye, or they may be embodied as a part of the thimble, which is interposed between the fittings and the eye to prevent any change in shape or size of the eye under strain. No solder whatever is used (Figs. 13-15). The effi-

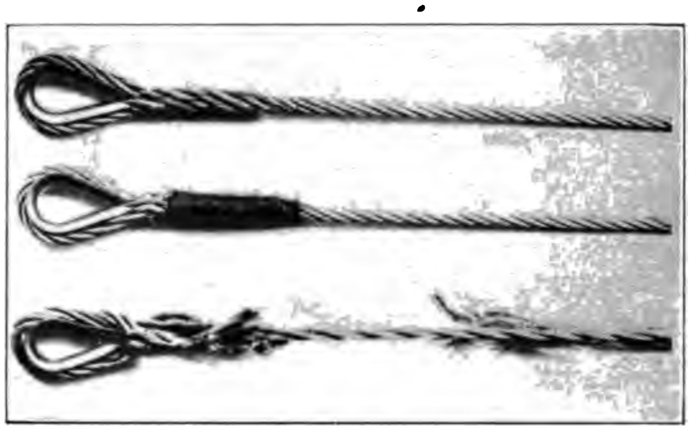


FIG. 16. SERVED AND UNSERVED SPLICED JOINTS AND TYPICAL FRACTURE IN AVIATOR CORD.

ciencies obtained with such terminals were very uniform, ranging only from 92 to 96 per cent, with an average of 94 per cent. Such a terminal, although necessarily somewhat complex, has marked advantages. It can readily be made up in the field, and there are unlikely to be hidden defects, any slipshod workmanship being instantly apparent on inspection.

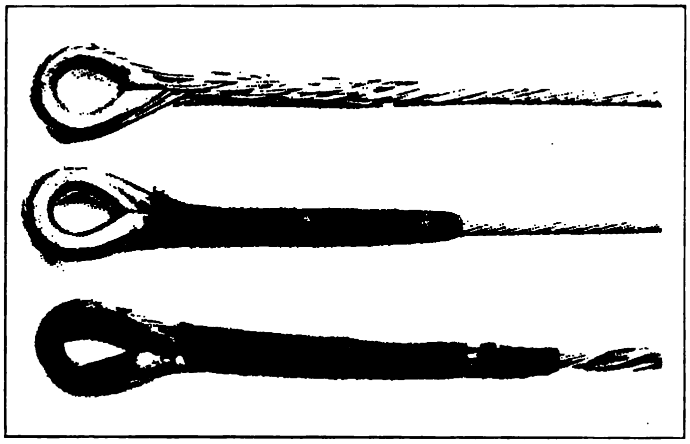


FIG. 17. SERVED AND UNSERVED SPLICED JOINTS AND TYPICAL FRACTURE IN AVIATOR STRAND.

The British standard, which has recently been adopted by the Society of Automotive Engineers, calls for the use of the plain wire coil ferrule with solder. A device which has been considerably used in England, although not yet employed in this country, is the streamline wire with swaged and threaded ends, thus doing away with the necessity for turnbuckles. Such wires are very expensive and difficult to make, but have decided advantages in practice. They are unlikely to come into use except for fighting machines, where cost is of no importance.

FIG. 18. UNSOLDERED FIELD TERMINAL FOR AVIATOR STRAND.

Cable Terminals

Both strand and cord can be spliced with excellent results if the work is done by an expert rigger. Roebling's tests indicated an efficiency of from 80 to 85 per cent for aviator cord with spliced and served terminals (Fig. 16), and from 90 to 100 per cent, the highest values corresponding to the smallest wire sizes, for 19-wire aviator strand (Fig. 17). The break always occurred at the last tuck in the splice, which would suggest the advisability of tapering the splice to a greater extent.

For field connections, fittings similar to those recommended for solid wire, consisting of a thimble embodying a wedge and a ferrule of soft wire (Fig. 18), gave excellent results, showing an efficiency of 90 per cent.

The status of solder is the same as in the case of solid wire. 100 per cent efficiencies can be secured by the use of thimble, ferrule, and solder with either strand or cord, but there is the same risk of injury to the material through improper manipulation.

In connection with the larger diameters of strand, sockets may advantageously be used (Fig. 19). They give 100 per cent efficiency, or very nearly; they require no high degree of skill to apply, and the fitting is neat and simple in appearance. The common type consists of a conical shell, the hole in the small end being just large enough to admit the strand. The strand is passed through this hole for a short distance, unravelled, and the ends spread out as much as possible. The conical shell is then poured full of solder, and the ends of the component wires cut off flush with the large end of the shell. The only danger in the use of such a fitting arises from the liability to deterioration of the solder.

As we mentioned in the preceding chapter of the course, the stress diagram which was then drawn does not form a complete basis for the choice of members, but should be supplemented by various other diagrams corresponding to different conditions of loading. We shall, therefore, confine ourselves to figuring, for purposes of illustration, a few of those members which are most heavily stressed under the conditions which we have already considered.

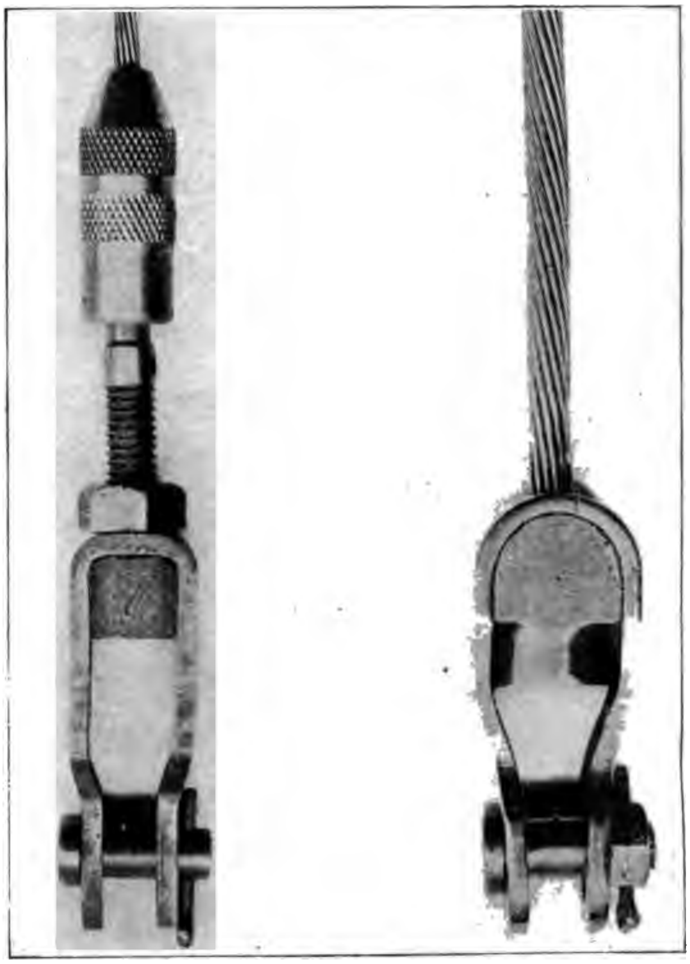
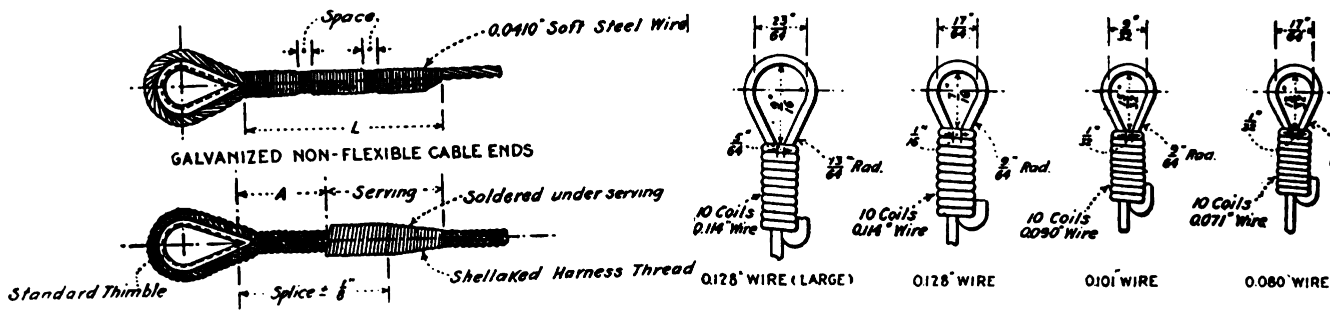


FIG. 19. SOCKET TERMINALS FOR AVIATOR STRAND.

Since a dynamic load factor of 8 has already been allowed for, we shall use a factor of safety above this of only one and a half. This is equivalent to an overall factor of safety, relative to the static load, of twelve, a value which is fairly representative of modern practice in the design of bodies for training machines. The latest specification issued by the Government calls for an overall factor of ten, but this relates to pursuit machines, which are to be flown by skilled pilots only, and in which the factor of safety is purposely kept low in order to make possible a better performance, and hence a higher degree of military safety. In the case of those portions of the longerons which are curved to a considerable extent



FIGS. 20, 21, 22. S. A. E. STANDARD TERMINALS FOR SOLID WIRE, STRAND, AND CORD.

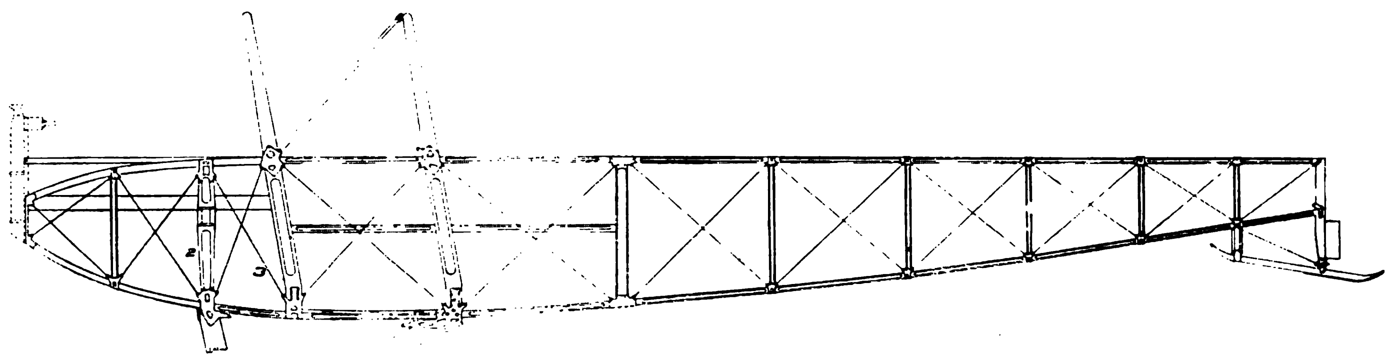


FIG. 23. LAYOUT OF BODY FOR STANDARD TRAINING MACHINE.

between struts, the factors of safety should be considerably increased, as a strut which has even the slightest sign of initial curvature will support much less load than one which is perfectly straight.

The members for which we shall compute the required size include a longeron section, a strut, and a wire, and they are marked with numbers on the layout drawing. We shall consider all compression members as perfectly square, although channeling is commonly employed, especially in struts and the rear portions of the longerons.

(1) The length of the section is 40 in. We shall try, as a first assumption, a section $1\frac{1}{4}$ in. square. The crippling load is then $17.44 \times 1{,}825{,}000 \times \frac{(1\frac{1}{4})^4}{40^2 \times 12}$, or 4,050 lb. This corresponds to a factor of safety of $\frac{4{,}050}{2{,}070}$, or 1.95, above the dynamic loading. We shall, therefore, use this section. It is well to have the factor in the longerons slightly greater than in the struts, since their end conditions approach less closely to fixation.

(2) The length of the strut is 31 in., and the compressive load is 3,900 lbs. Here, again, a section $1\frac{1}{4}$ in. square will be tentatively chosen. The crippling load equals $17.44 \times 1{,}825{,}000 \times \frac{(1\frac{1}{4})^4}{31^2 \times 12}$, or 6,740 lb. The factor of safety is then 1.72.

(3) The tensile load is 3,400 lbs. We shall select for this stay 19-wire strand $\frac{7}{32}$ in. in diameter. The breaking load of such strand is 6,100 lb., and the factor of safety, allowing for 90 per cent efficiency of the terminal connections, is 1.61.

The struts which carry the weight of the engine should be materially heavier than would be indicated by considerations of dead loading alone, since they are constantly submitted to a live, vibrative load, and, in addition, are subjected to bending forces because of the gyroscopic action in diving. These forces are calculable, but such an analysis is beyond the scope of this paper.

In Fig. 23 is shown the layout of the body. The only points at which channeled struts are used are the forward panels, which have the duty of transmitting the propeller thrust to the longerons, and thence to the wings. The other struts are made octagonal by chambering off the corners slightly. The longerons are channeled everywhere in back of the forward chassis strut, except that they are left solid for a few inches adjacent to every strut.

References for Part II, Chapter 10

First Annual Report of the National Advisory Committee on Aeronautics, Report No. 2; Government Printing Office, 1916.

Chapter XI

Wing Structure Analysis for Biplanes

There are many difficulties in the analysis of a biplane structure: the distribution of loading between upper and lower planes; the resolution of loading in the planes of the lift truss and the internal bracing; the resolution of loading to give bending moments on the spars, and the alternative methods which may be employed in drawing up the stress diagrams. But in the following notes is developed a system which is now generally employed, and which although it is not rigidly exact, gives sufficiently accurate results for practical needs, and as a system of comparison for machines which have been successful in flight.

Distribution Between Planes

The information available regarding distribution of loads

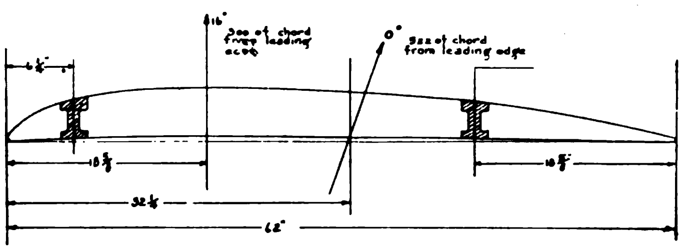


Fig. 1

between planes is scanty and contradictory. In practice it is sufficient to follow this equation:

(1) $W = (A_u x)\frac{11}{9} + A_l x$, where W = gross loading of the machine, A_u = area of upper wing, A_l = area of lower wing, x = gross loading per square foot on lower wing, $\frac{11}{9}x$ = gross loading per square foot on upper wing.

Unless the biplane truss falls away very much indeed from the conventional form, this will be a fair approximation.

Spacing of Wing Spars—Limiting Angles of Incidence

As the angle of incidence of a wing changes its center of pressure moves, and accordingly varying loads are placed on the rear and front spars (the center of pressure motion in a biplane is assumed to be identical with that of a monoplane). The spar spacing has to be so arranged that too great a proportion of the load is not thrown on either of the spars within the range of the usual angle of flight. This would be the case were the spars too close together or placed so that one of them would be quite close to one center of pressure. At the same time, the spars must not be placed too near either the front or the rear edge, so that there is always sufficient depth of spar. Thus in the machine the design of which we are carrying through, the spars are placed as shown in Fig. 1, about 10 per cent from leading edge and about 30 per cent from trailing edge, where the centers of pressure at 0 deg. and at 16 deg. are indicated. The loading is in this case

	Front spar	Rear spar
At 0°	29.8%	70.2%
At 16°	66.6%	33.3%

0 deg. and 16 deg. are taken in our design as the limiting angles of incidence, although very possibly the machine might fly both at some negative angle, and at some angle above 16 deg.

Running Loads

Applying equation (1) where W = 1793 lb. and A_u = 188 sq. ft., A_l = 175 sq. ft. for our machine, we find that the gross loading per square foot on the upper wing is 5.4 lb./sq. ft. and on the lower wing it is 4.43 lb./sq. foot. In the same manner the total gross weight supported by the upper wing is 1020 lb. and the total gross weight supported by the lower wing is 773 lb.

For simplicity, the running load is assumed to be uniform from tip to tip of the wings, and hence the gross running lifts are for a span of 36 ft. 6 in., 28.0 lb./foot on upper wing and 22.8 lb./foot on lower wing.

It is from the gross running lifts per foot that we obtain the running drifts per foot run, by dividing by the L/D ratio. Thus we have

	L/D	Upper wing running drift in lb./ft. run	Lower wing running drift in lb./ft. run
At 0	7.2	3.90	3.18
At 16°	6.92	4.05	3.30

Next it is necessary to determine the net running lift. To do this it is necessary to make assumptions for the weight of the wings and the weight of the interplane bracing.

Thus for the upper wing, assuming a weight of .73 lb./sq. ft., and half the weight of the interplane bracing of 91.5 lb. to be carried by it, we have a net lift of 1020 — 137 — 45.7 = 837.3 lb. or 22.9 lb./ft. run, and for the lower wing 773 — 128 — 45.7 = 599.3 lb. or 17.7 lb./ft. run.

We can now tabulate our results in such form that they can be used in resolving forces in planes of lift trussing and of the wings.

At 0° — Percentage front spar 29.8, Percentage rear spar 60.2

	Upper wing	Lower wing
Gross loading per foot run..	28 lb.	22.8 lb.
Drift per ft. run front spar.	1.16 lb.	.95 lb.
Drift per ft. run rear spar..	2.74 lb.	2.23 lb.
Net lift per ft. run front spar	6.85 lb.	5.27 lb.
Net lift per ft. run rear spar	16.05 lb.	12.43 lb.

At 16° — Percentage front spar 66.6, Percentage rear spar 33.3

	Upper wing	Lower wing
Drift per ft. run front spar..	2.70	2.17
Drift per ft. run rear spar..	1.35	1.09
Net lift per ft. run front spar	15.20	11.80
Net lift per ft. run rear spar	7.65	5.90

Resolution of Forces in Planes of Wing Trussing and of Wings, and in Plane of Spar Web

In Figs. 2 and 3 are shown the resolutions of forces at 0 deg. and 16 deg. respectively. It will be noticed that the resultant force in the plane of the lift truss is decomposed in plane of the spar web. It is this component in the plane of spar web

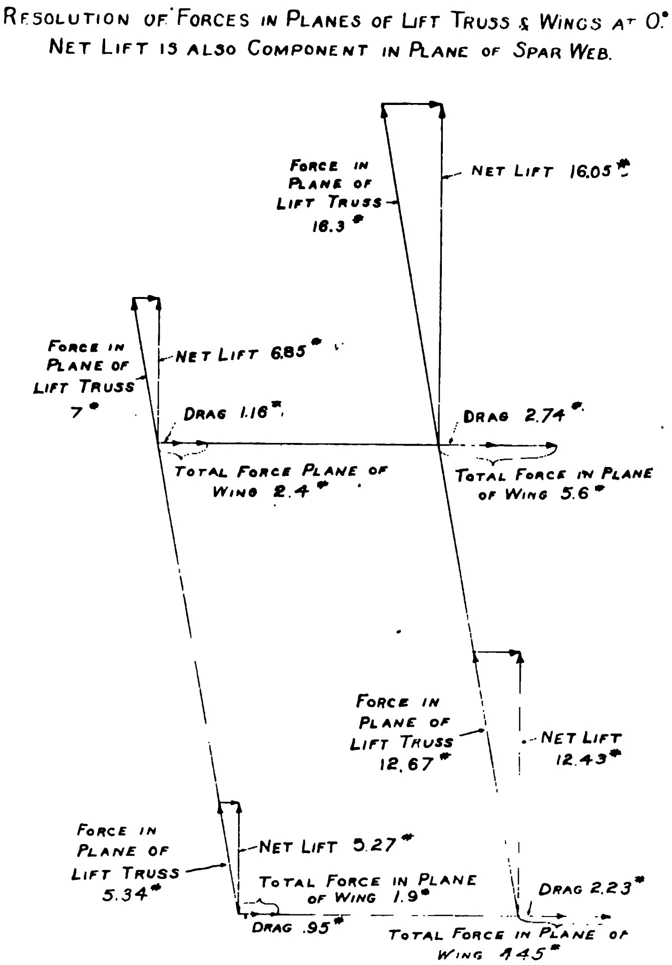


FIG. 2

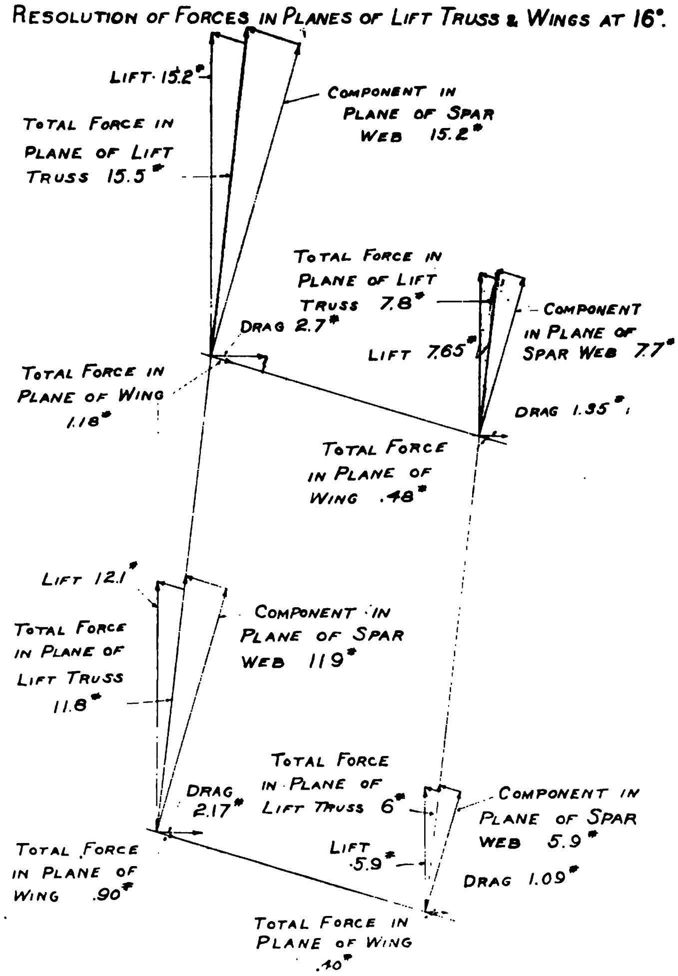


FIG. 3

which is subsequently used to draw the bending moment diagrams for the spars. This is a slightly arbitrary procedure. It would be more accurate to take the force in the plane of the lift truss as producing bending, but there would then be the complication of computing moments of inertia about an axis not perpendicular to the web.

From these resolutions it is now possible to tabulate figures which can be employed in the lift truss stress diagram, the drift bracing stress diagram, etc.

At 0°

	Upper wing. Front spar.	Upper wing. Rear spar.	Lower wing. Front spar.	Lower wing. Rear spar.
Force in plane of lift truss/running foot.	7 lb.	16.3 lb.	5.34 lb.	12.67 lb.
Force in plane of wing/running foot.	2.4 lb.	5.6 lb.	1.9 lb.	4.45 lb.
Force in plane of spar web/running foot.	6.85 lb.	16.05 lb.	5.27 lb.	12.43 lb.

At 16°

	Upper wing. Front spar.	Upper wing. Rear spar.	Lower wing. Front spar.	Lower wing. Rear spar.
Force in plane of lift truss/running foot.	15.5 lb.	7.8 lb.	12.1 lb.	6.0 lb.
Force in plane of wing/running foot.	1.18 lb.	0.48 lb.	0.90 lb.	0.40 lb.
Force in plane of spar web/running foot.	15.2 lb.	7.75 lb.	11.9 lb.	5.9 lb.

Figs. 2 and 3 indicate some peculiar results. Thus at 0 deg., part of the net lift is resolved into the plane of the wing, greatly increasing the demands on the internal wing bracing. Were the stagger of the biplane more pronounced, this effect would be still greater, and that is one of the disadvantages of excessive stagger. But at 16 deg., in this particular case, the component of the net lift along the plane of the wing relieves the internal wing bracing.

Different Methods Employed in Stress Diagrams for Lift Truss

Two distinct methods have been adopted in getting out stress diagrams for the lift truss.

(1) The trussing is treated as if pin jointed throughout by the ordinary bridge truss method, and the bending moments for the spars found as if they were freely supported at the ends, with uniformly distributed loads.

(2) The spars are treated as if continuous, so that bending moments in them and reactions at their supports are found by theorem of three moments. Then the reactions having been found, the stress diagram is drawn with such reactions as a basis.

The first method has the advantage of simplicity and of giving a very large factor of safety. The second method is much more difficult, but probably is nearer the mark, and we shall employ it accordingly.

Bending Moment Diagrams: Theorem of Three Moments

Any good text book on applied mechanics treats fully of the theorem of three moments, so that the following notes will be of the briefest:

In Fig. 4 is shown a beam loaded with unequal distributed loads over the two spans. At the three supports, 0, 1, 2 M_0, M_1, M_2 are corresponding bending moments; R_0, R_1, R_2 are corresponding reactions; S_{+0}, S_{-0}; S_{+1}, S_{-1}; S_{+2}, S_{-} are shears on either side of the supports.

If the beam is continuous over the three supports and has

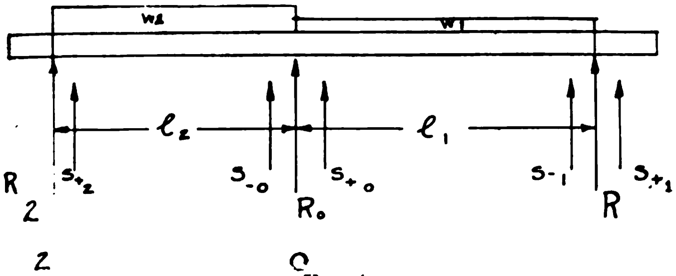


FIG. 4

the same cross-section throughout, the bending moments at the supports and the loads are connected by the following formula:

$$M_1l_1 + 2(l_1 + l_2)M_0 + M_2l_2 = -\frac{w_1l^3_1}{4} - \frac{w_2l^3_2}{4}.$$

All difficulties in working the theory of three moments are due to mistakes in the conventional signs.

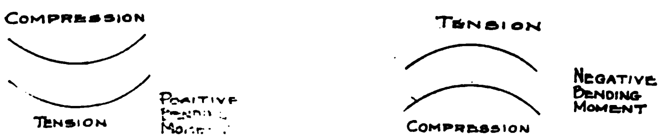


FIG. 5

The convention for bending moments is shown in Fig. 5. From this follows the rule:

Forces to left of a point must tend to turn a beam clockwise about that point in order to give a positive bending moment at that point—anti-clockwise to give a negative bending moment.

Forces to the right of a point must turn the beam anti-clockwise about that point in order to give a positive bending moment at that point—clockwise to give a negative bending moment.

If these rules are observed, the effect of the fixing moments is also automatically determined. Thus if a fixing moment is found to be negative at a support, and the above rules are followed, its effect will be negative on either side of that support.

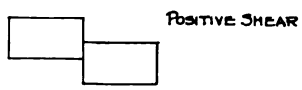


FIG. 6

The convention for shear is shown in Fig. 6. If forces to

FIG. 7

the left of a point tend to shear the beam upward, the shear at this point is positive. As a result of this arbitrary rule, when finding the shear by means of bending moments, the sign as found must be reversed if the origin chosen is to the left of the line of action of the shearing force.

Observance of this rule is not so important as the observance of the rule for bending moments. It is generally easier to see what happens physically.

Working Out of Bending Moment and Shear Diagrams for Upper Rear at 0 Deg.

The principles of the preceding paragraph will be best illustrated by working out the above case fully. In Fig. 7 is shown the disposition of the wing. With a total span of 36 ft. 6 in. and an engine panel of 2 ft. 6 in., we allow an overhang of 2 ft. 6 in., 7 ft. 9 in. for outer span between struts. and a smaller inner span of 6 ft. 9 in., which seems a reasonable spacing. The loading in plane of spar web as previously found is 16.05 lb./ft. run. For simplicity's sake, we neglect the engine panel.

To get bending moments at supports:

(a) $M_1 = O$, since wing is hinged at engine panel;

(b) $M_2 = \frac{16.05 \times (2.5)^2}{2} = 50.2$ lbs.-ft.;

(c) to find M_0 we write

$$M_1l_1 + 2(l_1 + l_2)M_0 + M_2l_2 = -\tfrac{1}{4}w_1l^3_1 - \tfrac{1}{4}w_2l^3_2$$

and by substituting in this equation,

$$M_0 = 93.7 \text{ lbs.-ft.}$$

To get shears at supports:

(a) $S_{-2} = 2.5 \times 16.05 = 40.1$ lbs.

(b) Taking moments about support 0 we have

$$M_2 + S_{+2}l_2 - \frac{wl^2_2}{2} = M_0;$$

therefore $S_{+2} = -56.7$ lbs.

(c) Taking moments about support 2 we have

$$M_0 - \frac{wl^2_2}{2} + S_{-0}l_2 = M_2;$$

therefore (reversing signs),

$$S_{-0} = 68.0 \text{ lbs.}$$

(d) Taking moments about support 1 we have

$$M_0 - \frac{wl^2_1}{2} + S_{+0}l_1 = M_1;$$

therefore $S_{+0} = -63.3$ lbs.

(e) Taking moments about support 0 we have

$$M_1 - \frac{wl^2_1}{2} + S_{-1}l_1 = M_0;$$

therefore (reversing signs)

$$S_{-1} = 40.4 \text{ lbs.}$$

To find the total reactions (the absolute sum of the shears), we have

$$S_{-2} + S_{+2} = R_2 = 96.8 \text{ lbs.}$$
$$S_{-0} + S_{+0} = R_0 = 136.3 \text{ lbs.}$$
$$S_{-1} = R_1 = 40.4 \text{ lbs.}$$

After having found the bending moments, shears, and reactions at the supports, it is very easy to draw the entire bending moment diagram by finding points of zero shear and maximum bending moment.

Thus in the outer span, if x is the distance to the right of support 2 of the point of zero shear, $S_{+2} = xw_2$ and $x = \frac{56.7}{16.05} = 3.53$ ft.

The bending moment at this point is (taking forces to the left)

$$M'_2 = \frac{wx^2}{2} + S_{+2}x = -49.8 \text{ lbs.-ft.}$$

Similarly in the inner span, if x = distance to the right of support 0 of the point of zero shear,

$$S_{+0} = xw_1,$$

and $x = 4.26$ ft.

The bending moment at this point is (taking forces to the left)

$$M'_0 = \frac{wx^2}{2} + S_{+0}x = -51.3 \text{ lbs.-ft.}$$

References for Part II, Chapter 11

"Wing Data and Analysis for a Staggered Biplane," by Dr. A. F. Zahm, Franklin Institute, December 1914.

British Report 1912-1913. No. 83. A preliminary note on methods of calculation which may be employed in the determination of the stresses in the spars of airplane wings by Bairstow and MacLachlan.

Chapter XII

Wing Structure Analysis for Biplanes

Reactions in Plane of Lift Truss Due to Upper Rear Spar at 0 Degree

At the conclusion of the previous chapter, we drew the bending moment diagram for the upper rear spar as a continuous beam, and found the appropriate reactions. But since the bending moment diagram was drawn for that component of the force in the plane of the lift truss which was in the plane of the spar web, allowance has to be made for it on reverting to the lift truss. The running loads were in the ratio of 16.3 to 16.05. Hence reactions are

$$R_2 = 96.8 \times \frac{16.3}{16.05} = 98.3 \text{ lb.}$$

$$R_0 = 136.3 \times \frac{16.3}{16.05} = 138.4 \text{ lb.}$$

$$R_1 = 34.6 \times \frac{16.3}{16.05} = 41.0 \text{ lb.}$$

Reactions in Plane of Lift Truss Due to Lower Rear Spar at 0 Degree

Since the spacing of the supporting points on the lower wing is identical with that of the upper wing, and the slight overhang is the same, the bending moment diagram and the shears and reactions will be in direct ratio to the loads. The ratio of loads on upper plane to lower plane is 14 to 11.5. Hence reactions are

$$R_2 = 98.3 \times \frac{12.67}{16.3} = 75.4 \text{ lb.}$$

$$R_0 = 138.4 \times \frac{12.67}{16.3} = 107.5 \text{ lb.}$$

$$R_1 = 41.0 \times \frac{12.67}{16.3} = 31.4 \text{ lb.}$$

Stress Diagram for Rear Lift Truss at 0 Degree

We are now in a position to draw the stress diagram for the lift truss as shown in Fig. 1. The only other load to be added is 20.3 lb., which is allowance for half the air force due to the engine panel acting on the rear spar.

In drawing this stress diagram, the strut $K\,L$ is assumed as taking no tensile load, and the lift load at $F\,G$ is transmitted by the cross wire $L\,M$ to the body.

Stress Diagram for Internal Upper Wing Bracing at 0 Degree

In Fig. 2 is drawn the stress diagram for the internal bracing of the upper wing at 0 deg. incidence.

The spars have so much less resisting moment in the plane of the wing that it is perfectly justifiable to treat the interplane wing bracing as a pin-jointed structure and neglect all consideration of bending moments.

The running loads per foot run are taken from the preced-

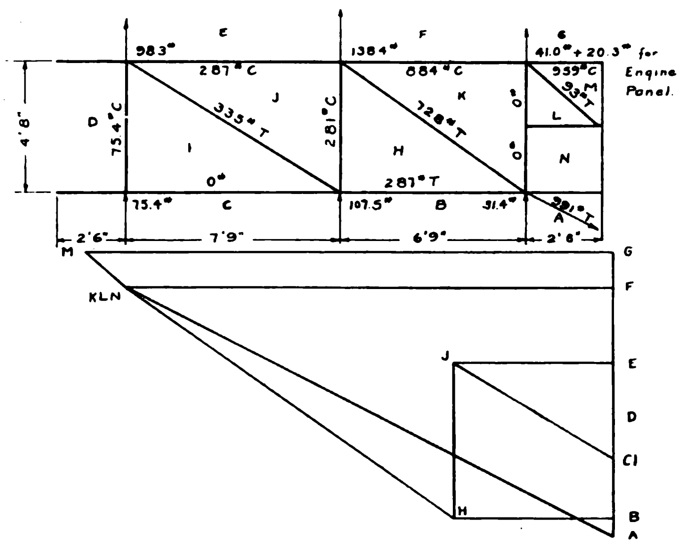


FIG. 1 STRESS DIAGRAM OF REAR LIFT TRUSS AT 0 DEG. INCIDENCE

ing sections, with the addition of ½ lb. drift at each external bracing point.

Computations for Dimensions of Rear Upper Spar

Having drawn the bending moment diagram, the lift truss stress diagram and the internal wing bracing stress diagram,

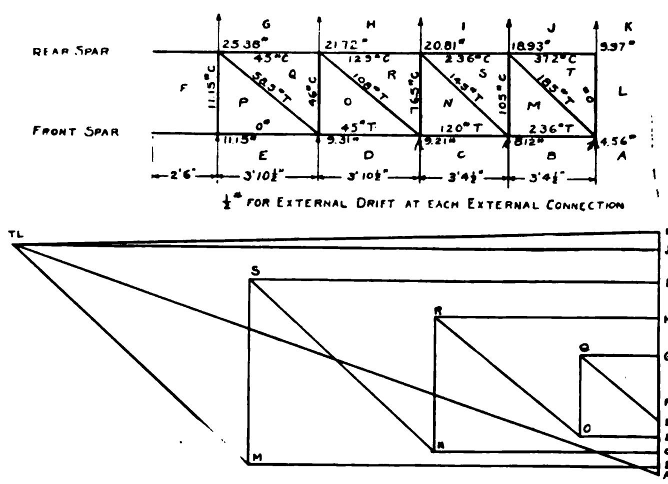


FIG. 2 STRESS DIAGRAM OF UPPER WING INTERNAL DRAG BRACING AT 0 DEG. INCIDENCE

all at 0 deg. incidence, we are in position to determine the dimensions of the rear upper spar. Since the worst loads come on the rear spar at this angle of incidence, it is not necessary to recompute it at 16 deg. also.

The worst loads it has to meet occur in the inner span, 3 feet from the wing hinge:

Compression from the lift diagram of 675 lb.

Compression from the drag diagram of 330 lb.

Bending moment of 44 ft. lb.

It is first of all necessary to fix the effective depth of spar for the wing section employed, namely, the R.A.F.6. The spar is placed at 30 per cent from the rear edge, where the thickness of the wing is .054 of the chord. For a 62-in. chord, this gives a thickness of 3.34 in., or 3 21/64 in. very nearly.

From this must be deducted the thickness of the two rib caps or flanges. The construction and dimensioning of ribs is a matter of some uncertainty and controversy, and will be dealt with fully in a later article. We will assume for the moment that a thickness of ⅛ in. will be sufficient for the rib flanges, so that the effective depth of the flanges will be reduced to 3 5/64 in. The actual drawing up of the beams is largely a matter of trial and error. That is to say, an apparently suitable section has to be drawn in its area, and moments of inertia, etc., have to be computed together with the factor of safety consequent thereon.

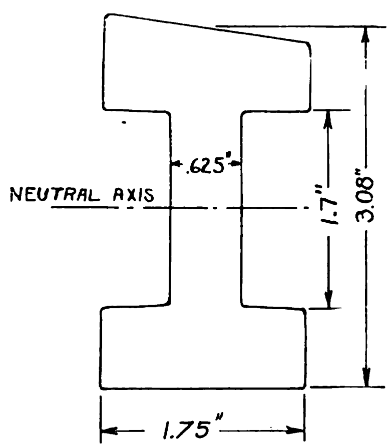


Fig. 3 Rear Upper Section

After a number of trials, the spar section of Fig. 3 is found to be satisfactory.

The upper surface of the spar follows the outline of the R.A.F.6 wing section at this point, but in making computations the slight slope may be neglected.

To compute the moment of inertia, the quickest way is to compute for the solid section and deduct the moment of inertia of the material channeled out. The moment of inertia of a rectangle being given by the formula $\frac{bd^3}{12}$

$$I = \frac{1.75 \times 3.08^3}{12} - \frac{1.125 \times 1.7^3}{12} = 4.27 - 0.46 = 3.81$$

$$A = 1.75 \times (3.08) - 1.125 \times 1.7 = 5.39 - 1.91 = 3.48$$

The stress in the outermost fibers will now be given by the formula

$f = \frac{P}{A} + \frac{My}{I}$ where P = direct load, y = distance of outer fibers from the neutral axis and f = stress. Since $P = 1120$ lb., $M = 93.7$ ft. lb. $= 1125$ inch. lb. and maximum $f = \frac{1120}{3.48} + \frac{1125 \times 1.54}{3.81} = 777$ lb.

Allowing a maximum fiber stress for spruce of 6500 lb., we get a factor of safety of 8.35, which is in excess of the 7.5 specified by the Army.

Similar computations can be carried out for the upper front spar at 16 deg.—since the biggest load is carried at this angle.

It must be pointed out, however, that although the formula $f = \frac{P}{A} + \frac{My}{I}$ is largely used, and is, therefore, perfectly sound on a comparative basis, the factor of safety given by it is not exactly true. Tests on breaking beams by bending show great variations from the above formula, depending largely on sections employed, but special values for moduli of rupture by bending are not available.

For the lower wing, if the same chord is employed as in the upper wing, and the spars have the same dimensions, no computations need be made, since the loads on the lower wing will always be considerably less. Whether with the same chord the lower spars should be smaller than the upper ones is a matter to be determined largely from the manufacturing point of view.

A Complete Example of Wing Analysis Arrangement

In Fig. 4 is shown the complete analysis for the wing structure of a Curtiss biplane. The methods employed in getting out this analysis are substantially the same as indicated above, and the method of presentation is an excellent model.

Computations for Shear in Spars

Wood is so much weaker in shear than in either tension or compression, that it is somewhat surprising that designers do not make computation for shear in the spar web—although spars are always made solid for 2 in. or 3 in. on either side of a supporting point, to allow for the maximum shear occurring at such points.

The maximum longitudinal shear for a beam which is subjected to vertical shear occurs at the neutral axis, and its value is determined by the formula

$$q = \frac{F}{Ib} A_1 y$$

where F = vertical shear at the point due to external loads, I = moment of inertia of whole section, b = breadth of web at neutral axis, A_1 = area of section above neutral axis, y = distance of centroid of this area from the neutral axis.

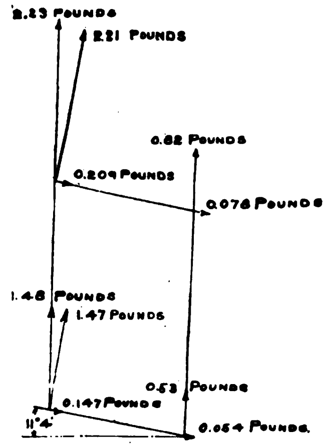
Thus consider the same upper rear spar 6 inches from support O. The shear at this point, as given by the shear force diagram of Fig. 7 of the preceding chapter, is 60 lb. Considering the section of spar shown in Fig. 3:

$$I = 3.81 \text{ in.}^4$$
$$A_1 = 1.76 \text{ in.}^2$$
$$b = 0.625 \text{ in.}$$
$$y = 0.93 \text{ in.}$$
$$q = \frac{60}{3.81 \times 0.625} \times 1.76 = 0.93 = 41.2 \text{ lb.}$$

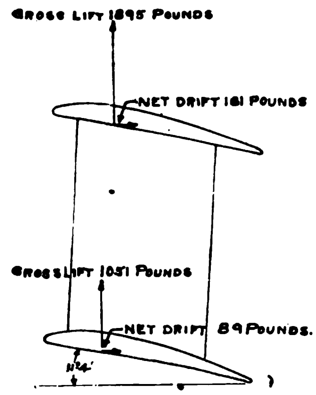
Allowing shearing value of spruce to be 400 lb./sq. inch, we have a factor of safety of 9.7, which is amply sufficient. But cases might occur where the shear near supports is very large, and resistance to shear being largely due to the web, it is always advisable to make such computations.

Fig. 4

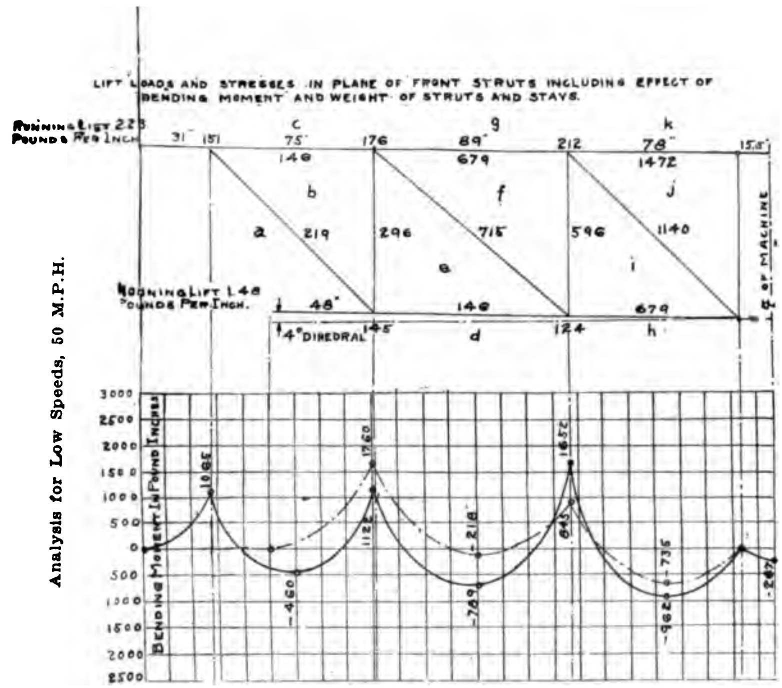
Analysis for Low Speeds, 50 M.P.H.



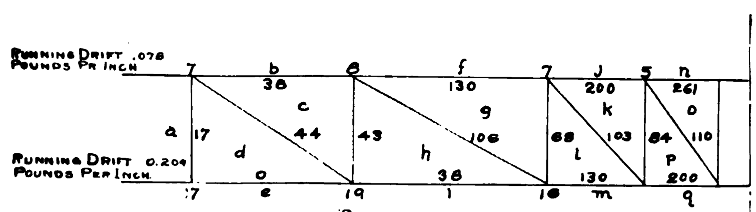
NET RUNNING LIFT AND DRIFT COMPONENTS
PLANES OF WINGTRUSSING



GROSS LIFT AND NET DRIFT
ON SUPPORTING PLANES

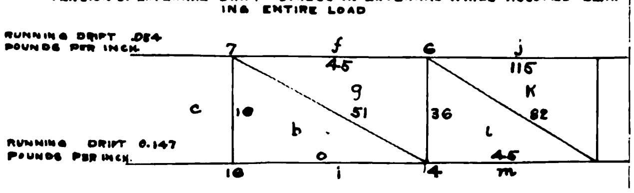


BENDING MOMENTS ON FRONT, TOP AND LOWER SPARS
IN PLANE OF I BEAM WEB.



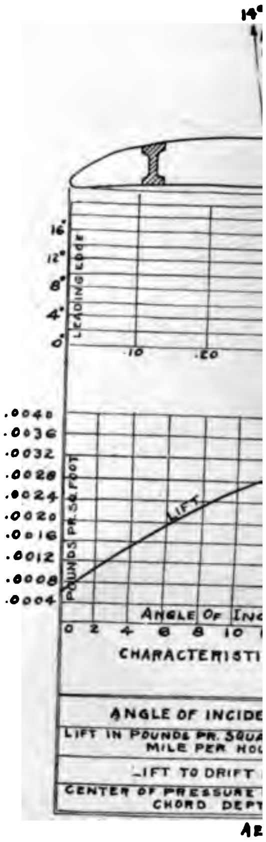
PLAN OF UPPER SPARS

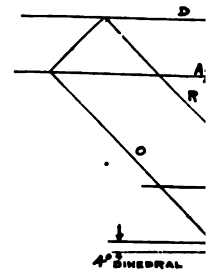
DRIFT LOADS AND STRESSES IN THE PLANES OF THE UPPER AND LOWER SPARS INCLUDING EFFECT OF STRUT AND WIRE RESISTANCE AND EXCLUDING TENSION OF EXTERNAL DRIFT STRESS IN EXTERNAL WIRES ASSUMED BEARING ENTIRE LOAD



PLAN OF LOWER SPARS.

FIG. 4



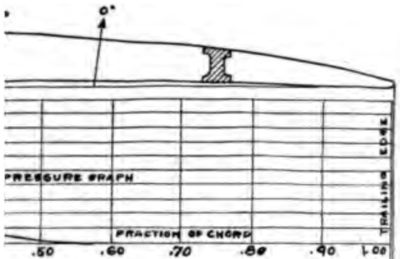


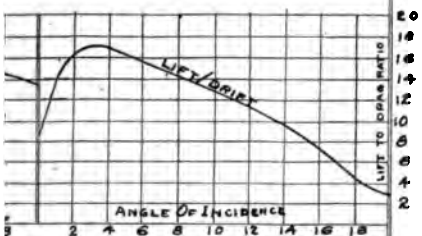
SYMBOL	MEMBER	MATERIAL	SECTIONAL AREA SQUARE INCHES	MOMENT OF INERTIA	SECT
A	FRONT SPAR	SPRUCE	4.0	8.1	3.
B	"	"	"	"	"
C	"	"	"	"	"
D	REAR SPAR	"	4.73	5.48	3.5
E	"	"	"	"	"
F	"	"	"	"	"
G	FRONT SPAR	"	4.0	8.1	3.
H	"	"	"	"	"
I	REAR SPAR	"	4.73	5.48	3.5
J	"	"	"	"	"
K	LIFT STRUT		4.88	0.79	
L	"		7.88	2.07	
M	"		4.16	0.57	
N	"		6.5	1.39	
	LIFT CABLE	SOFT ROEBLING STEEL	NO. OF CABLES	DIA. OF CABLES	BREA LOA
O	"	"	2	5/32	64
P	"	"	2	"	"
Q	"	"	2	7/32	1120
R	"	"	2	5/32	64
S	"	"	2	"	"
T	"	"	2	7/32	112

FACTORS OF SAFETY ARE
FOR STRESS IN DRIFT WIRE

ASSEMBLED DAT

SULTANT WING FORCE AT VARIOUS ANGLES
INCIDENCE AND SAME SPEED

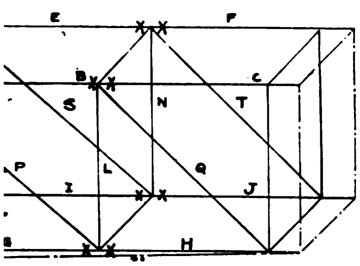




MONOPLANE WING SHAPE - AIR AT 15° CENT.
ASPECT RATIO 6/1

	0°	2°	4°	6°	8°	10°	12°	14°	16°	18°
NE	.0009	.0011	.0015	.0018	.0022	.0025	.0028	.0030	.0030	.0029
	6.5	16.3	17.1	15.6	14.2	12.6	11.6	9.7	7.6	4.3
F	.575	.425	.358	.329	.312	.302	.292	.280	.296	.330

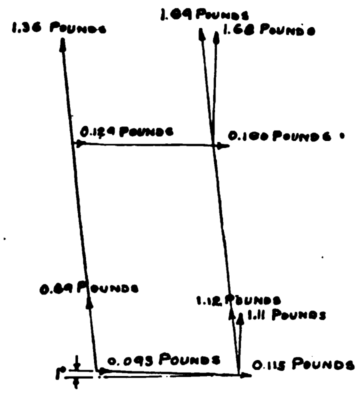
DATA AND DIAGRAMS



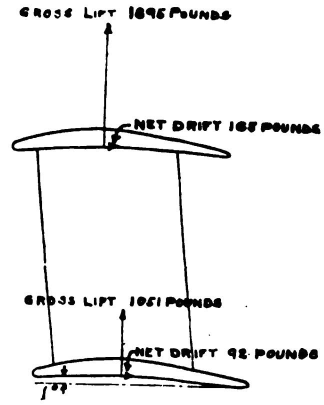
OF GREATEST STRESS MARKED THUS "X"

Low Speed			For High Speed			
ESS IN LBS./SQ.IN.		Factor of Safety	Fibre Stress in Lbs./Sq. In.			Factor of Safety
Lift	Drift		Bending	Lift	Drift	
-37	0	20	±181	-22	0	33
-170	+9	11	±266	-104	+13	19
-369	+50	9	±266	-226	+70	16
-12	-8	49	±241	-24	-10	24
-53	-28	26	±356	-109	-38	13
-115	-55	19	±356	-236	-80	10
+37	+0	13	±273	+22	+0	23
+170	+11	16	±149	+104	+15	25
+12	-9	30	±365	+24	-13	18
+53	-24	58	±182	+109	-36	26
-296		8		-181		14
-596		11		-364		18
-109		17		-224		8
-221		20		-452		10
+219		29		+134		47
+715		9		+436		14
1140		9		+696		16
+81		79		+166		38
+265		24		+542		11
+421		26		+865		13
OR POINTS OF GREATEST STRESS STRUTS SEE DRIFT DIAGRAMS.						

COMPUTED VALUES

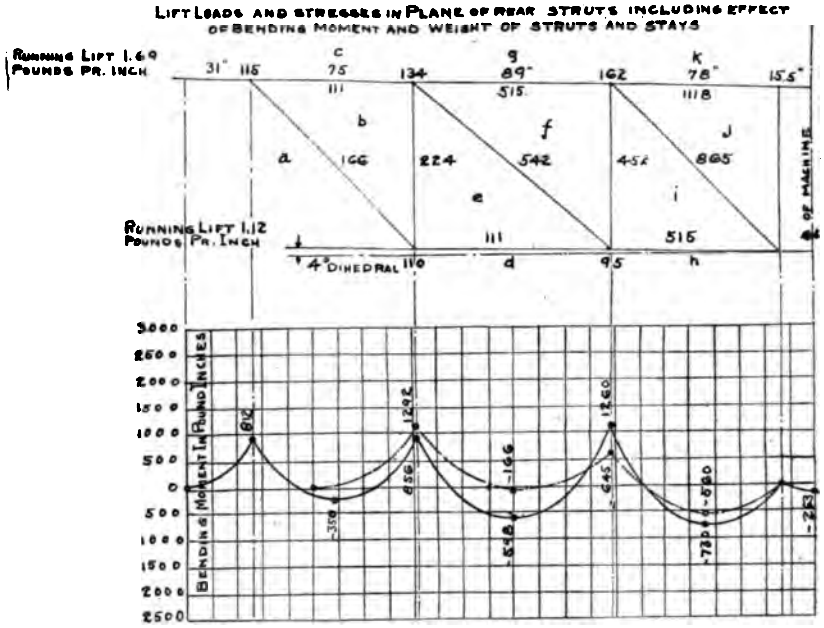


Net Running Lift and Drift Components
in Planes of Wing Trussing

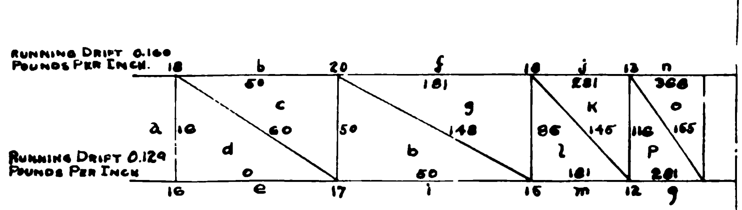


Gross Lift And Drift
on Supporting Planes

Lift Loads and Stresses in Plane of Rear Struts Including Effect of Bending Moment and Weight of Struts and Stays

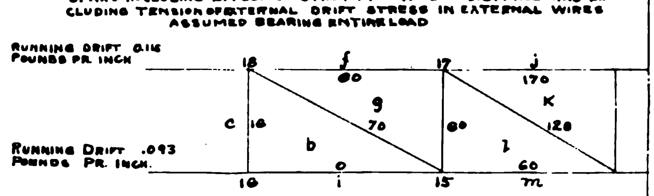


Bending Moments on Front, Top and Lower Spars
in Planes of I Beam Web.



Plan of Upper Spars

Drift Loads and Stresses in the Plan of the Upper and Lower Spars Including Effect of Strut and Wire Resistance and Excluding Tension of External Drift Stress in External Wires Assumed Bearing Entire Load



Plan of Lower Spars.

Appendix

Notes of Aerial Propellers

By H. Bolas

Presented by Mervyn O'Gorman, Superintendent of the Royal Aircraft Factory

Reports and Memoranda, No. 65. March, 1912

The object of the present notes is to give an account of a method which has been employed in propeller design at the Royal Aircraft Factory, with some particulars as to the theoretical assumptions on which it is based. In principle the method is essentially the same as that which has been described by M. Drzewiecki,* and is commonly referred to as the constant incidence method.

Constant Incidence Method.—In this method the propeller blade is regarded as an aerofoil, each element of which makes a constant angle with its path in space. In other words, the blade is treated exactly as though it were an airplane, except that the path of the blade is a helix instead of a straight line.

The path in space of any point of a propeller moving forward with constant velocity is a helix, the advance of the screw per revolution being called the pitch. If the angle of the blade at any point corresponds with that of the effective helix, the only resistance to motion is head resistance and skin friction, and no thrust is obtained. If, however, any element is set at some angle of incidence to the effective helix it becomes an aerofoil possessing lift and drift, and a propeller so designed will give a definite thrust.

In this method, then, the angles of the effective helix are first calculated for various fixed points of the blade for a given velocity of advance and a given propeller speed. Each of these angles is then augmented by the angle of attack.

Value of Angle of Attack.—The value of the angle of attack to be used depends chiefly on the form of blade profile adopted.

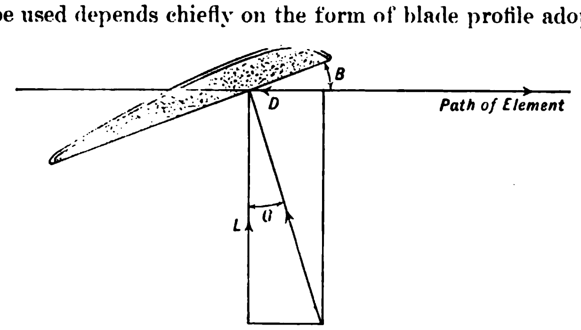


FIG. 1

Consider Fig. 1, which is intended to represent a section of an aerofoil, or of a propeller blade, in motion. D = drift or resistance to motion; L = lift (not to be confused with thrust of element); B = angle of attack; G = gliding angle, or ratio D/L nearly. As the angle of attack B is varied the ratio D/L will also vary, and for some particular value of B this ratio will be a minimum. It is this value of B which, on this method, is employed, so far as possible, in practice. As already stated, this best value depends on the form of blade profile; it is usually found to be in the neighborhood of 4° † for good forms. It is well to note, however, that in some cases it is impossible to use this best value, as the width of blade required may be too great. In such a case a compromise must be made and the angle of attack increased. This only happens, however, when the primary conditions are bad.

Efficiency of Elemental Strip,‡ and Curve of Efficiencies

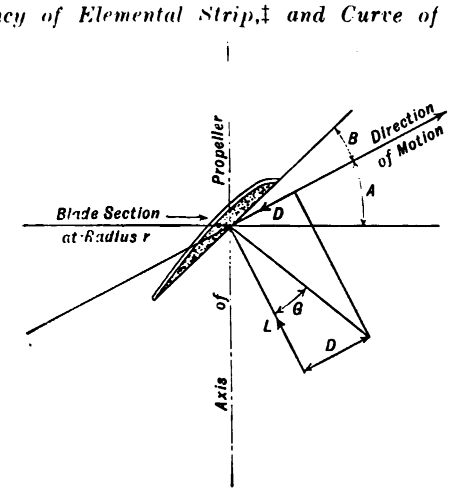


FIG. 2

(see Fig. 2).—Let A denote the angle the effective helix makes with the line at right angles to the axis, B the angle of attack. Also as above, let $D/L = K = \tan G$. Then the efficiency E of the element is readily shown to be given by

$$E = \frac{\tan A}{\tan(A+G)}$$

and for variation of G, E is a maximum when G is a minimum.

From this a curve of efficiencies for different values of the angle A can be plotted (Fig. 3). The curve in Fig. 3 has been

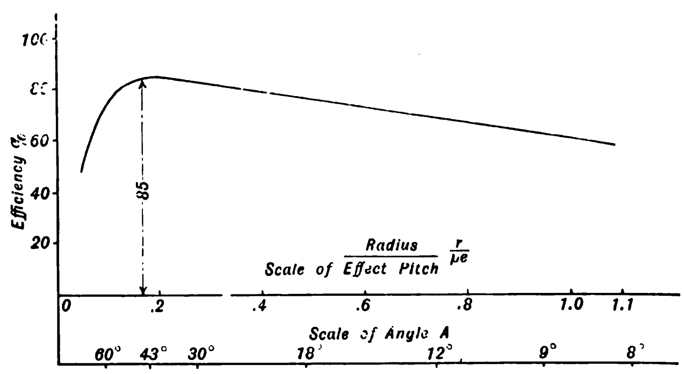


FIG. 3

drawn for $G = 4° 35'$. In propeller diagrams it is more convenient to employ r/p_e as abscissa instead of the angle A

* See Abstracts No. 45, Report for 1909-10. *Theorie Generale des Propulseurs Helicoidaux et Methode de Calcul de ces Propulseurs pour l'Air.* Paris, F. Louis Vivien, 1909.

† See Notes added June, 1912, p. 136.

‡ Defined as the efficiency of a strip of blade at radius r and of width δr this strip being supposed *not* isolated from the neighboring elements.

where p is the effective pitch, the relation between them being

$$r/p_e = \frac{1}{2\pi}.\cot A.$$

One other point may be mentioned—the question of *variation* of efficiency as we travel out along the blade. It will be seen from the curve, that as we increase r/p_e the efficiency increases very rapidly at first, and reaches a maximum at the point where $r/p_e = 0.17$ (or $A =$ about 43°). From the formula given above for E it is readily shown that E becomes a maximum where $A = 45° - (G/2)$. Thus in the particular case given, $G = 4° 35'$, and the angle for max. efficiency $= 42° 43'$. As r/p_e increases still further, it will be noticed that the efficiency decreases and continues to do so. Hence, beyond a certain radius on the blade the elements get less and less efficient towards the tip. This is an unfortunate point in the design of actual propellers.

Propeller Diagrams.—Supposing the velocity of advance, the revs. per minute of the propeller, and the horse-power of the engine are known, we can now proceed to set out our

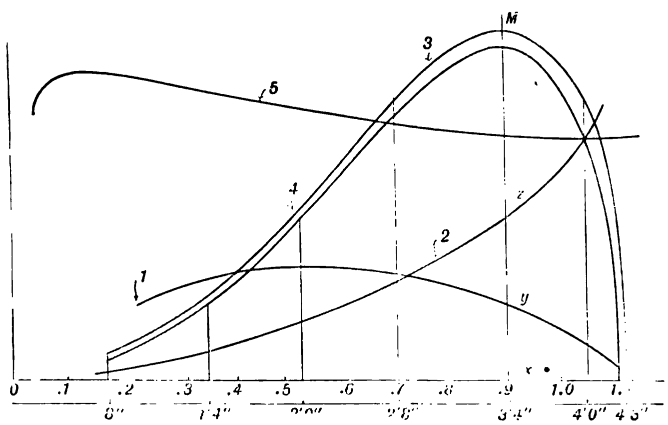


FIG. 4

diagrams. The scales of these are in the first place immaterial. (*See* Fig. 4.) It will be best to explain first the meaning of the various curves, and afterwards to devote some attention to the ideals to be aimed at and the determination of scale.

Curve 1.—This is termed the *Linear Grading Curve*, and its ordinates are everywhere proportional to the blade widths, these being supposed to be developed on to the plane of the paper.

Curve 2. Pressure per sq. ft. curve.—The pressure upon the blade per unit area of surface at any point depends principally upon (i) the form of section, (ii) the angle of attack, (iii) the velocity of that particular point relative to the air. Let C be a constant depending upon the form of section. Let B as before be the angle of attack, and V_t the velocity of a particular element relative to the air. Then we may write with sufficient truth—

$$\text{Pressure per sq. ft.} = CBV_t^2,$$

and since C and B are constants along the blade we can write

$$\text{Pressure per sq. ft.} \propto V_t^2.$$

Now if V be the axial velocity of translation

$$V_t^2 = V^2/\sin^2 A.$$

But V^2 also is constant along the blade. We may therefore put

$$V_t^2 \propto 1/\sin^2 A,$$

and it is now only necessary to plot a curve, the ordinates of which are proportional to $1/\sin^2 A$, in order to obtain the pressure per sq. ft. diagram. The scale is for the present immaterial.

Curve 3. Load Grading Curve.—Consider any value of r/p_e represented by Ox. Then at this point the width of blade is represented by xy. Evidently then the quantity $(xy \times xz)$ will be a measure of the load per foot run on the blade, and if we perform this operation for a number of points at different radii we can draw a curve, the ordinates of which will represent the loading per foot run. Curve 3 (Fig. 4) has been obtained in this way.

Curve 4. Thrust Grading Curve.—The thrust grading curve is such that the ordinate at any point represents the thrust per unit length (per foot) of blade, and is obtained from the load grading curve in the following manner.

Consider the sketch shown in Fig. 5, which represents a sec-

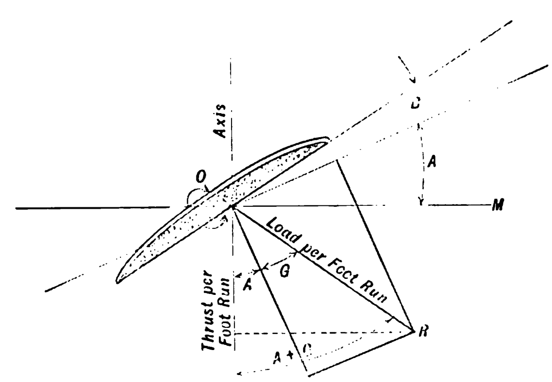


FIG. 5

tion of the blade at any radius. The line RO, which represents the pressure per unit length upon the element, makes with the axis (or direction of thrust) an angle $(A + G)$, where as before A is the angle the effective helix makes with OM, and G is the gliding angle. Both A and G are known for any point of the blade. Now we have:—

Thrust per foot run = load per foot run $\times \cos(A + G)$.

The ordinates of the thrust grading curve are thus obtained from those of the load grading curve by multiplying by $\cos(A + G)$.

Curve 5. Efficiency Curve.—The values of the angle A and gliding angle G being known, the efficiency at any point is given by

$$\frac{\tan A}{\tan(A + G)}$$

and an efficiency curve can be plotted as described earlier.

It should be explained that the order in which we set out the diagrams will depend upon our initial data. For instance, if we are given the shape of the thrust grading diagram, we may first lay down the load grading diagram, then the pressure per square foot curve, and finally, from the previous two, the linear grading curve, viz., the plan form of the blade. On the other hand, if we start with the plan form curve, we may,

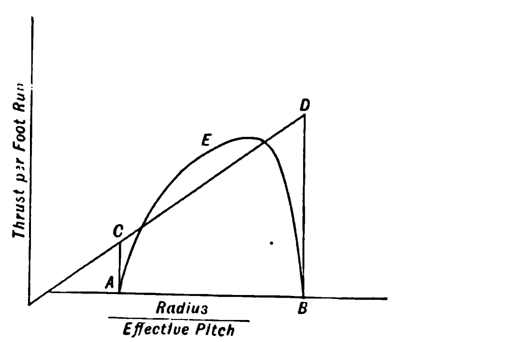


FIG. 6

by reversing the above process, finally arrive at the thrust grading curve. In practice the latter method is always adopted for reasons which we are now in a position to explain.

Ideal Curves.—If we assume as an ideal condition that the velocity in the slip stream is everywhere parallel to the axis and uniform, then the momentum per second imparted, and hence the thrust at any radius, will be proportional to that radius. In other words, the ideal thrust grading diagram is a straight line passing through the origin, as shown in Fig. 6—*ACDB*. In practice, however, such a form of diagram would be undesirable, even if attainable, and some compromise as that sketched in Fig. 6—*AEB*—would have to be adopted. According to Mr. Lanchester, the best practical shape of thrust grading curve is that shown in the next figure, in which "conjugate"* points on the diagram have equal efficiencies (Fig. 7).

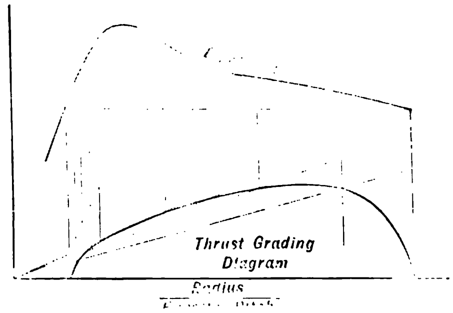


FIG. 7

If, however, we started out with a diagram of this type, and from it constructed a linear grading curve, our final plan form would take the shape shown in Fig. 8. Such a blade could

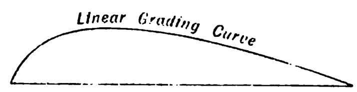


FIG. 8

never be employed in practice, since under existing conditions we should require an immense blade width and a very large diameter in order to obtain the needed thrust. It is useful, however, inasmuch as we know the direction in which to work when given good conditions at the start.

When designing then, as I have already stated, we invariably begin with our plan form, and finish up by obtaining a thrust grading diagram, which will usually differ considerably in shape from the ideal diagram first described.

The propeller curves being laid down, it only remains now to give them their proper scales in order that we may satisfy the initial requirements.

Determination of Scales.—Since the ordinates of the thrust grading diagram are measures of the thrust per foot run along the blade, and the abscissae represent the radii in feet (p, the effective pitch being constant), it will be evident that the *area* of the thrust grading diagram represents to some scale the total thrust on the blade.

$$\text{Now thrust per blade } t = \frac{T}{\text{No. of blades}},$$

where T is the total thrust of the propeller.

Let p = horizontal scale (known), viz., 1 inch on diagram = p ft. of radius.

Let q = vertical square (required), viz., 1 inch on diagram = q lbs. per ft.

Then $pq \times$ area of diagram in square inches = Thrust per blade.

Hence

$$q = \frac{\text{Thrust per blade}}{\text{area of diagram} \cdot p}.$$

In this equation the horizontal scale p is known, the area of the diagram may easily be computed by means of a planimeter, and it is then only necessary to find t, the thrust per blade, in order to determine completely the vertical scale.

Before t can be calculated, however, the total efficiency of the propeller must be found (see Fig. 9). To do this we divide up our thrust grading diagram into a number of parts and

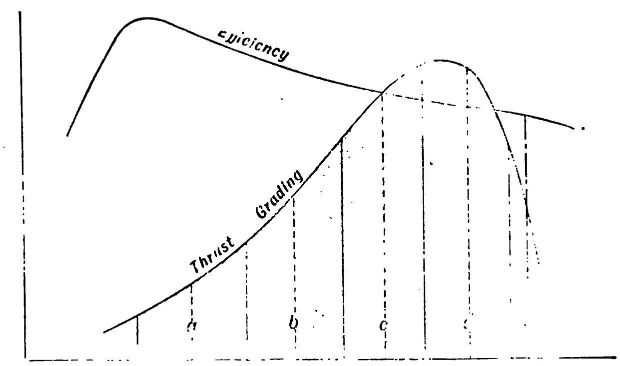


FIG. 9

then compute the area of each. The mean efficiency of each part is now read off on the efficiency curve, then divided into its corresponding area, and all the quotients so obtained are summed up. The sum arrived at in this way, divided into the area of the thrust grading diagram, will give the total efficiency of the blade.

Computation of Thrust.—Let H = H.P. of engine, E_t = total efficiency of propeller, V = velocity of advance in ft. per sec., T = thrust of propeller.

Then $$TV/550 = H.E$$

or $$T = H \times E_t \times 550/V.$$

We then have

$$\text{Thrust per blade} = T/\text{No. of blades}.$$

The thrust per blade having been thus ascertained, the vertical scale of the thrust grading diagram is calculated as before explained, from

$$q = \frac{\text{Thrust per blade}}{\text{area of diagram} \times p}.$$

Thus in a given case $H = 58$, $E = .67$, $V = 73$ ft./sec.

Therefore

$$T = \frac{58 \times .67 \times 550}{.73} = 292 \text{ lbs.}$$

No. of blades = 4.

Therefore, thrust per blade = 73 lbs.

Further, area of thrust grading diagram = 40 sq. ins.

$$p = 0.381 \text{ ft.}$$

Therefore

$$q = \frac{73}{40 \times 0.381} = 4.79 \text{ lbs. per ft.}$$

Load Grading Curve.—Since this was obtained by dividing the ordinates of the thrust grading curve by cos $(A + G)$, which is itself a mere ratio (and has therefore no dimensions), the scales, both thrust and load grading curves, will evidently be identical.

Pressure per sq. ft. Curve.—The determination of the intensity of pressure on the blade at any point is of course a matter for experiment, and the data at present available are somewhat scanty. In an account of the recent experiments of M. Eiffel, however, a curve will be found which gives the lift and drift for a particular form of section, and this form of section is the one we have adopted. A rough* reproduction of M. Eiffel's curve is shown in Fig. 10, some explantion of which is perhaps necessary.

The angles of incidence (viz., angles of chord UV) are marked along the curve itself. Consider the point where the

* Conjugate points are defined as the points in which a straight line through the origin cuts the thrust grading diagram.

* NOTE.—This curve is to be taken as diagrammatic only.

angle is 4° (P) and let the ordinate there be denoted by $K'y$ and the abscissa by $K'x$. Then we have for this particular angle of attack—

Lift per unit area of surface $= K'y \times (\text{velocity})^2$.
Drift " " $= K'x \times$ "

Further, if the point P be joined to O, the angle POY is the

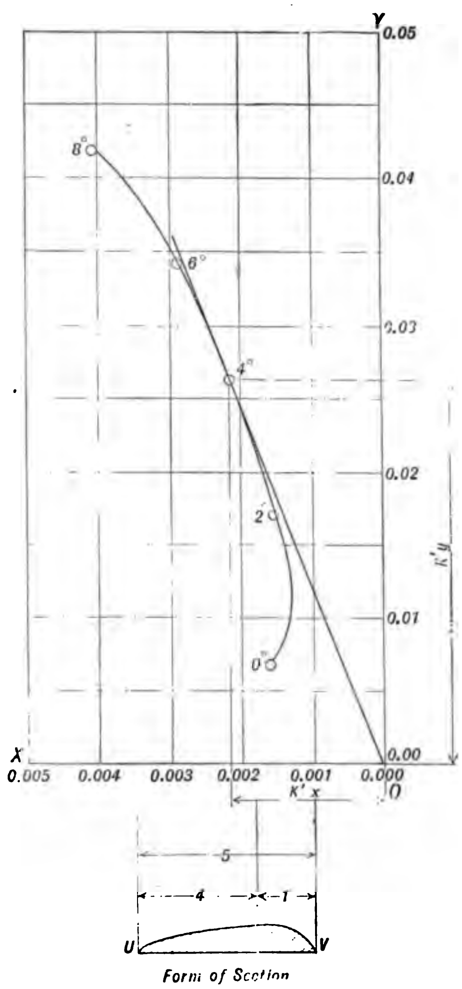


FIG. 10

gliding angle of the aerofoil, since the tangent of POY is the ratio of drift to lift of the plane. It will therefore be evident that the best angle of incidence is obtained by drawing a tangent from O to the curve, since this gives the least value of drift/lift, and what is more, it will be seen that the best angle for this particular section is 4°. With this preliminary explanation we can now proceed to the scale of our diagram: choose some point on the blade, say the point X where $r/p_e = 0.9$.

Then as before explained,

$$\cot A = \frac{2\pi r}{p_e}; \text{ whence } A = 10°\text{-}2',$$
$$\text{and } \sin A = 0.1742.$$

But we have previously shown that

Absolute velocity of point = Velocity of advance/sin A.
= 420 ft. per sec.

Now from Eiffel's curve, for angle of attack = 4°, $K_y =$ 0.0005* (in English units).

Therefore, lift in lbs. per sq. ft. $= K_y$ (velocity in ft. per sec.)^2
$= .0005\,(420)^2 = 88.$

* K_y (English units) = Eiffel's constant ÷ 50.

NOTE.—Numerical examples to be treated as indicating method only. See notes added June, 1912, p. 185.

Again, we have from the pressure per sq. ft. curve—

$XZ = 3''.42$
$= 88$ lbs. per sq. ft.

Therefore $1'' = 25.7$ lbs. per sq. ft.,

thus fixing the scale of our pressure per sq. ft. curve.

It will be noticed in the above that I have taken actual *lifts* on blade instead of *total* pressure, viz., $\sqrt{(Ky^2 + Kx^2)}.V^2$, but the difference is usually so small as to be negligible.

Linear Grading Curve.—Again consider the point X of this, where $r/p_e = .9$

We have

$$\left\{\begin{matrix}\text{Ordinate of blade}\\ \text{in feet at point}\end{matrix}\right\} \times \left\{\begin{matrix}\text{pressure per}\\ \text{sq. ft. in lbs.}\end{matrix}\right\} = \text{load per ft. run of blade.}$$

$$\therefore \quad \text{Ordinate of blade in ft. at point} = \frac{\text{load per ft. run}}{\text{pressure per sq. ft.}}$$
$$= \frac{XM \times 4.79}{XZ \times 25.7} = 0.4 \text{ ft.}$$

$XM = 7''.33$
$XZ = 3''.42$
$\therefore X = 1''.7 = 0'.4$ of blade width.
$\therefore 1'' = 0'.235$ of blade width

We have also

Max. blade width $= 0'.557 = 6''.8$, say $7''$ wide.

The scale of the linear grading curve being known, we are now in a position to set out our propellers, since the blade angles at the various radii have been previously determined.

Number of Blades.—At the present time, the majority of propellers in use are of wood, and hence two or four blades are employed, three blades being excluded for constructional reasons.

It is difficult at present, until further experimental data are available, to decide as to the relative merits of two- and four-bladed propellers. The four-bladed propeller is possibly better aerodynamically and from the point of view of balance, but the two-bladed propeller involves much less work in construction and is stronger at the boss.

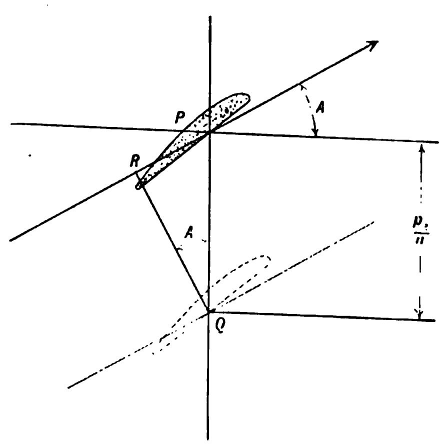


FIG. 11

Interference.—Interference, or the disturbing action which any one blade exerts upon the air dealt with by any other, is a matter about which little is yet known. The following suggestion is put forward merely as affording a rough guide in design.

Reverting to comparison with the airplane, moving in a continuous straight line, we may look upon the blades of a propeller as superposed aerofoils which travel in a helical path. Assume that the thickness of air stratum affected by the blade is a constant proportion of the blade width at any point. Then we may write (Fig. 11)

$$RQ = PQ \cos A = \frac{p_e}{n} \cos A.$$
$$(n = \text{No. of blades}).$$

Let blade width at point $= b$, and put, according to above assumption, $\frac{RQ}{b} = m$.

Then b must be less than $\frac{p_e}{m \times n} \cos A$.

If we now plot a curve where abscissa represent radii, or r/p_e, and whose ordinates are the calculated values of $\frac{p_e}{m \times n}$. $\cos A$, we shall arrive at what we call the ***limit curve***, and the linear curve should at all points lie within this if there is to be no interference. Such a curve has been plotted for the case of the propeller already mentioned, and is shown in Fig. 12.

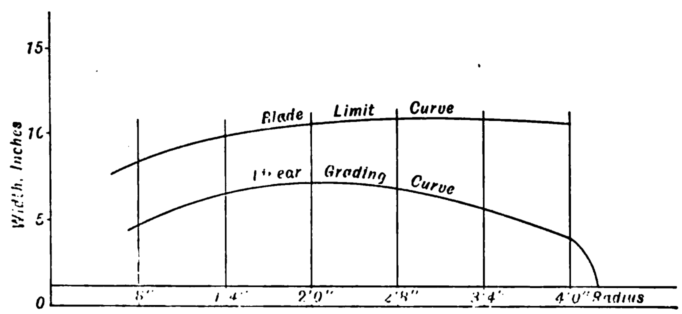


FIG. 12

Construction and Strength of Blade.—Having now indicated the method of fixing the sizes of a propeller in order to satisfy given mechanical and aerodynamical conditions, it would appear desirable to devote a little attention to the actual constructional design. The following remarks are made with reference to the usual type of wood propeller, though the manner of procedure is quite general whatever the material adopted. The process is essentially one of trial and error. The extreme radius of the blade being known, a number of sections are decided upon, say 6 or 8 inches apart, and the blade angles at these points computed. The linear grading curve will now provide us with the necessary blade widths, and it is only necessary to set these down at their proper projection in order to determine the *true* plan form. The distribution of the width, however, about a line through the centre of the blade root has yet to be discussed. It is usual in design so to shape the blade that twisting action is either greatly minimized, or eliminated altogether, and for this reason a symmetrical plan form is undesirable. Fig. 13 will explain this point.

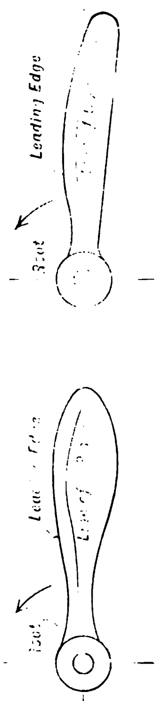


FIG. 13

A number of preliminary trial blade sections must now be sketched out, and previous examples of similar design will act as a good guide as to the thicknesses required. The next point is to estimate the strength of the blade, and the stresses to which this is subjected must be divided into (1) Centrifugal, (2) Bending. These are to be treated separately and then added together.

(1) *Centrifugal Stresses.*—It has been found convenient to write out the calculations in column form as follows:

No. of section.	Name of element.	Average radius.	Area of section.	Length of element.	Mean area.	Vol. of element.	Weight at .00123 lbs./cu.cm.	Centrifugal force on element.	Centrifugal force on section.	Centrifugal stress on section.
		feet.	sq. cm.	cm.	sq. cm.	cu. cm.	lbs.	lbs.	lbs.	lbs. sq. inch.
1	A	1.0	44	20.3	44	893	1.1	635	4245	620
2	B	1.67	44	20.3	40.5	822	1.01	970	3610	530
3	C	2.33	37	20.3	31.5	640	0.79	1060	2640	460
4	D	3.00	26	20.3	21	426	0.52	895	1580	390
5	E	3.67	16	20.3	11	223	0.28	590	685	275
6	F	4.125	6	7.6	4	31	0.04	95	95	102
							3.74	4245		

NOTE.—In computing weight of elements, cm, $(cm)^2$, and $(cm)^3$ were employed. This is merely a matter of convenience, the weights being obtained in lbs. and stresses calculated in lbs. per square inch.

Bending Stresses.—(*See* Figs. 14, 15, 16.)

In order to determine the bending stress, a bending moment

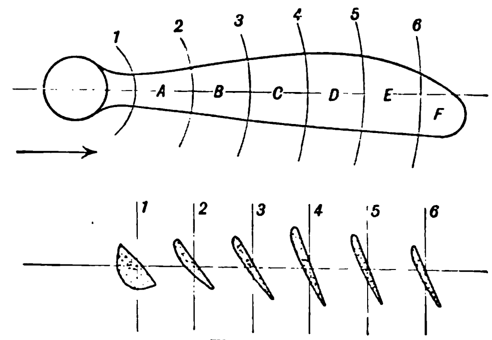


FIG. 14

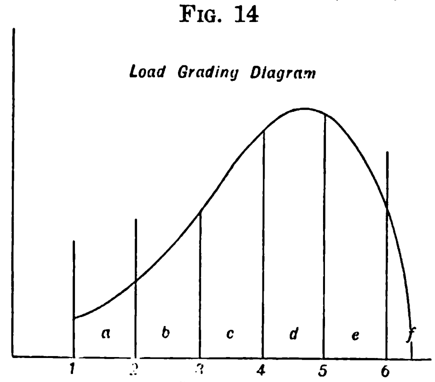


FIG. 15

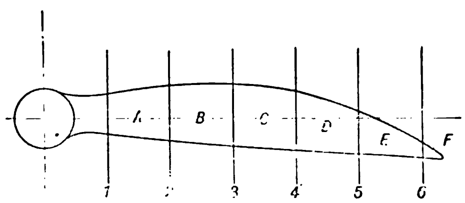


FIG. 16

diagram for the blade is first drawn, and it will usually be found good enough to assume all the loading uniplanar. Taking each of the elements A, B, C, D, E and F, estimate the load on each from the load grading diagram thus:

No. of section.	Name of element.	Area of portion of load-grading diagram, sq. inches.	Load expressed in lbs. for one blade.	Value of r/p_e at C.G. of portion.	Value of r in inches.	Distance from section in inches.	Bending moment. inch-lbs.	Bending stress on section. lbs. sq. in.
1	A	1.40	2.56	0.275	12.5	4.5	12	1150
2	B	4.34	7.95	0.45	20.5	12.5	99	1000
3	C	8.06	14.75	0.62	29.2	21.2	313	1050
4	D	11.78	21.55	0.79	36.0	28.0	604	1080
5	E	11.78	21.55	0.97	44.3	36.3	783	1000
—	F	2.94	5.38	1.07	48.8	48.8	220	—
	Total	49.3	73.74	—	—	—	2031	—